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Development and Utilization of Composite Honeycomb and Solid Laminate Reference Standards for Aircraft Inspections

Dennis Roach and Kirk Rackow

Prepared by
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Development and Utilization of Composite Honeycomb and Solid Laminate Reference Standards for Aircraft Inspections*

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Abstract

The FAA's Airworthiness Assurance NDI Validation Center, in conjunction with the Commercial Aircraft Composite Repair Committee, developed a set of composite reference standards to be used in NDT equipment calibration for accomplishment of damage assessment and post-repair inspection of all commercial aircraft composites. In this program, a series of NDI tests on a matrix of composite aircraft structures and prototype reference standards were completed in order to minimize the number of standards needed to carry out composite inspections on aircraft. Two tasks, related to composite laminates and non-metallic composite honeycomb configurations, were addressed. A suite of 64 honeycomb panels, representing the bounding conditions of honeycomb construction on aircraft, was inspected using a wide array of NDI techniques. An analysis of the resulting data determined the variables that play a key role in setting up NDT equipment. This has resulted in a set of minimum honeycomb NDI reference standards that include these key variables. A sequence of subsequent tests determined that this minimum honeycomb reference standard set is able to fully support inspections over the full range of honeycomb construction scenarios found on commercial aircraft. In the solid composite laminate arena, G11 Phenolic was identified as a good generic solid laminate reference standard material. Testing determined matches in key velocity and acoustic impedance properties, as well as, low attenuation relative to carbon laminates. Furthermore, comparisons of resonance testing response curves from the G11 Phenolic NDI reference standard was very similar to the resonance response curves measured on the existing carbon and fiberglass laminates. NDI data shows that this material should work for both pulse-echo (velocity-based) and resonance (acoustic impedance-based) inspections.

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This program is sponsored by the FAA William J. Hughes Technical Center under the direction of the technical monitor Dave Galella. The approach used in this effort was formulated and carried out by the Commercial Aircraft Composite Repair Committee (CACRC) Inspection Task Group which consisted of the team members listed below. The data contained in this report was produced by this project team.

CACRC Inspection Task Group
Gerry Doetkott, Northwest Airlines
Tom Dreher, United Airlines
Bruce Garbett, Airbus Industries
John Hewitt, Airbus Industries
Jim Hofer, Boeing
Jeff Kollgaard, Boeing
Nick Shaw, Boeing
Glae McDonald, US Airways
Kirk Rackow, Sandia Labs AANC
Dennis Roach, Sandia Labs AANC

At the Sandia Labs AANC, additional data acquisition was conducted by Phil Walkington and the statistical variance analysis of the NDI data was performed by Floyd Spencer. In addition, Airbus Industries partners conducted their own in-house validation testing to evaluate the standard's use on Airbus aircraft. Key participants include: Dieter Schiller (Airbus - Germany), Gunter Wehmann (Airbus - Germany), Jean-Baptiste Gambini (Airbus - France), Vicente Cortes (Airbus - Spain), Yolanda de Frutos (Airbus - Spain), Steve Spokes (Airbus -UK), and Rob Rose (Airbus -UK). Finally, FAA oversight has been provided by Al Broz, Chief Scientist and FAA National Resource Specialist in Nondestructive Evaluation and Fred Sobeck, FAA National Resource Specialist in Aging Aircraft.

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NDI Validation Results and Industry Adoption Via Aerospace Recommended Practices

EXECUTIVE SUMMARY

The rapidly increasing use of composites on commercial airplanes coupled with the potential for economic savings associated with their use in aircraft structures means that the demand for composite materials technology will continue to increase. Inspecting these composite structures is a critical element in assuring their continued airworthiness. The FAA recognizes a need to produce guidance that will assure the airworthiness of composite structures. The FAA's Airworthiness Assurance NDI Validation Center (AANC) operated by Sandia National Labs, in conjunction with the Commercial Aircraft Composite Repair Committee (CACRC), has developed a set of optimum composite reference standards to be used in nondestructive testing (NDT) equipment calibration prior to inspection of commercial aircraft composites.

The standards contain damage that is representative of that found in the field and include typical flaw scenarios such as disbonds and delaminations. Furthermore, this activity produced a workable number of reference specimens. Currently, the recognized number of variables makes the potential number of standards very large and unmanageable. Inspection characterizations and equipment responses were used to determine the important variables needed in a composite reference standard thus eliminating unnecessary standard configurations.

The main goal of this project was to develop standards that will allow for repeatable, accurate inspections. Many composite inspections are performed by visual inspections and tap tests. Composite inspection requirements are increasing and may soon surpass the capabilities of the tap test. This effort will aid the composite inspection process through the use of engineered reference standards and the utilization of more sensitive NDT equipment.

The project tasks addressed both composite laminates and composite honeycomb configurations. Through the active participation of the aircraft Original Equipment Manufacturers (OEMs), this project represents a harmonized approach taken by aircraft manufacturers in order to advance the interests of the airlines and OEM's worldwide. The airlines also worked together to support this project since it will aid their inspection practices and minimize the number of reference standards that they have to purchase and maintain.

The end result of this project was a set of composite calibration standards (laminate and honeycomb) to be used in NDT equipment calibration for accomplishment of damage assessment and post-repair inspection of all commercial aircraft composites. The CACRC-sited advantages of industry accepted composite standards include: 1) providing a consistent approach to composite inspection thus helping to minimize false calls, 2) reducing standard procurement costs, and 3) aiding the assessment of composite inspection technologies.

<u>Honeycomb Standard Activity</u> - In this project, the AANC conducted a series of NDI tests on a matrix of composite aircraft structures and prototype reference standards in order to: 1) minimize the number of standards needed to carry out composite inspections on aircraft, and 2) optimize

the inspections for maximum sensitivity and flaw detection. A suite of 64 honeycomb panels, representing the bounding conditions of honeycomb construction on aircraft, was inspected using a wide array of NDI techniques. An analysis of the resulting data determined the variables that play a key role in setting up NDT equipment. This resulted in an optimized set of minimum honeycomb reference standards that include these key variables. A sequence of subsequent tests determined that this minimum honeycomb reference standard set is able to fully support inspections over the full range of honeycomb construction scenarios. Supporting tasks established the best methods for engineering realistic flaws into the specimens. An Aerospace Recommended Practice (ARP) was produced under the auspices of the Commercial Aircraft Composite Repair Committee. This ARP is the vehicle by which the honeycomb standards are being adopted by the aviation industry worldwide.

Solid Laminate Activity - Part II of this document focuses on the solid laminate activity. The Commercial Aircraft Composite Repair Committee (CACRC) Inspection Task Group identified a need for a set of "generic" composite reference standards for use by operators in setting up their inspection equipment. The goal of this effort was to establish a single, generic composite laminate reference standard that will accommodate inspections on the full array of fiberglass and carbon laminates found on aircraft. Through-transmission ultrasonics was applied to the series of existing Boeing, Douglas, and Airbus laminate standards in order to measure the key velocity, acoustic impedance, and attenuation characteristics in the laminates. A material search identified an excellent, generic solid laminate reference standard material: G11 phenolic. Prototype laminate standards were fabricated from the G11 material. This report documents the validation data accumulated on the G11 standards to validate their use in aircraft inspections. A second Aerospace Recommended Practice (ARP) was produced to support industry adoption of the G11 laminate reference standards.

PART I: COMPOSITE HONEYCOMB NDI REFERENCE STANDARDS

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PART I: COMPOSITE HONEYCOMB NDI REFERENCE STANDARDS

1. INTRODUCTION

The aircraft industry continues to increase its use of composite materials, most noteworthy in the arena of principle structural elements. The extreme damage tolerance and high strength-to-weight ratio of composites have motivated designers to expand the role of fiberglass and carbon graphite in aircraft structures. This has placed greater emphasis on the development of improved nondestructive inspection (NDI) methods that are more reliable and sensitive than conventional NDI. The majority of composite honeycomb structure inspections are performed visually and supplemented by tap test methods. Tap testing, which uses a human-detected change in acoustic response to locate flaws, and more sophisticated nondestructive inspection (NDI) methods such as ultrasonics or thermography, have been applied to an increasing number of applications to detect voids, disbonds, and delaminations in adhesively bonded composite aircraft parts. Low frequency bond testing and mechanical impedance analysis tests are often used to inspect thicker laminates.

In 1991, the Federal Aviation Administration (FAA) established an Airworthiness Assurance Center at Sandia National Laboratories. Its primary mission is to support technology development, validation, and transfer to industry in order to enhance airworthiness and improve the aircraft maintenance practices of the U.S. commercial aviation industry. The rapidly increasing use of composites on commercial airplanes coupled with the potential for economic savings associated with their use in aircraft structures means that the demand for composite materials technology will continue to increase. Inspecting these composite structures is a critical element in assuring their continued airworthiness. To address these needs, the AANC, in conjunction with the Commercial Aircraft Composite Repair Committee, developed a set of composite reference standards to be used in NDT equipment calibration for accomplishment of damage assessment and post-repair inspection of all commercial aircraft composites. In this program, a series of NDI tests on a matrix of composite aircraft structures and prototype reference standards were completed in order to minimize the number of standards needed to carry out composite inspections on aircraft.

A suite of 64 honeycomb panels, representing the bounding conditions of honeycomb construction on aircraft, were inspected using a wide array of NDI techniques. An analysis of the resulting data determined the variables that play a key role in setting up NDT equipment. This has resulted in a prototype set of minimum honeycomb reference standards that include these key variables. A sequence of subsequent tests determined that this minimum honeycomb reference standard set is able to fully support inspections over the full range of honeycomb construction scenarios.

1.1 PURPOSE

The purpose of this project was to develop a set of composite calibration standards to be used in NDT equipment calibration for accomplishment of damage assessment and post-repair inspection of all commercial aircraft composites. The advantages of industry accepted composite standards include: 1) providing a consistent approach to composite inspection thus helping to minimize false calls, 2) reducing standard procurement costs by minimizing the number of reference standards that airlines have to purchase and maintain, 3) producing improvements in current composite inspection practices, and 4) aiding the assessment of advanced composite inspection technologies.

The goal of this project was to develop standards which will allow for repeatable, accurate inspections. The standards contain damage that is representative of that found in the field. The important variables needed in composite honeycomb reference standards have been determined via inspection characterizations and a study of NDT equipment responses. Finally, this project has introduced the use of various NDT devices to inspection applications on composite structures. Other inspection improvements will come through the development of optimized procedures and practices.

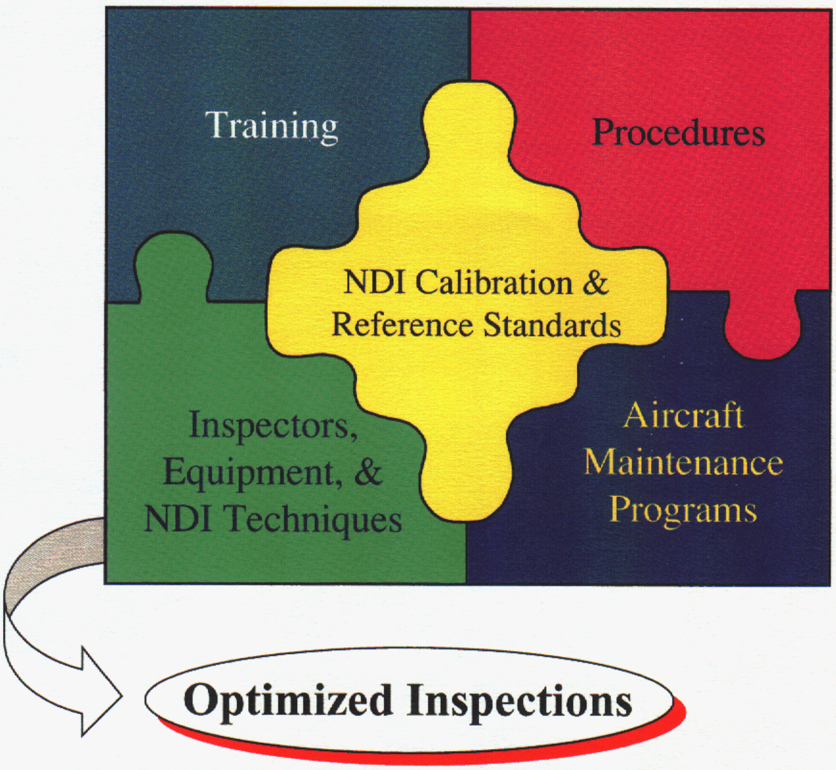

Figure 1: Factors Affecting Aircraft Inspections and Central Role of NDI Reference Standards

Current damage detection inspections are supported by reference standards that may be limited because they don't represent the actual structure. There may be a corresponding loss of confidence in the standards. In addition, most inspections are performed by the present inspection criteria which call for visual inspections and tap tests. Inspection requirements associated with the use of advanced composite materials are increasing and may soon surpass the capabilities of the tap test. This effort will aid the composite inspection process through the use of engineered reference standards and the utilization of associated NDT equipment. Also, there are no repair inspection reference standards. Repair inspections performed now are based on comparisons with undamaged, adjacent structure and an expected uniformity of response across a repair. This approach may result in a lack of consistency and/or false calls. Figure 1 summarizes the central role that composite reference standards play in achieving optimized inspection results.

1.2 COMPOSITE HONEYCOMB STANDARDS BACKGROUND

Composite materials are increasingly becoming the material of choice for aircraft designers because of their global benefits. Engineers estimate that building comparable fuselages with aluminum would take thousands of components and fasteners, and require extensive tooling and dozens of technicians. An aircraft would weigh about 20 percent more and consume more fuel. Through the use of composite technology construction, engineers can cut the number of parts in an assembly in half. This results in significant cost savings. Other benefits of composite technology include lower acquisition costs, lower operating costs, as well as improved maintainability, reliability and durability.

The capability of inspection techniques to detect flaws in composite structures must keep pace with the expanding use of composites on commercial aircraft structures. Figure 2 highlights the wide range of composite structures on commercial aircraft. New transport and commuter category aircraft, such as the Boeing 7E7 and the Airbus A380, are being produced with a majority of their structure composed of composite materials. Typical damage encountered in composite structures includes: 1) disbonds and delaminations stemming from normal flight loads, 2) fluid ingress, 3) impact damage, 4) lightning strikes, 5) deterioration from contact with fluids such as paint strippers or hydraulic fluids, and 6) extreme heat and ultraviolet exposure. Each of these elements can produce hidden damage that may be difficult to detect yet significantly detrimental to the strength of the structure.

After developing a Composite Inspection Handbook [1], the CACRC Inspection Task Group identified a need for a set of "generic" composite reference standards for use by operators in setting up their inspection equipment. The reference standards should include typical composite flaw scenarios - delaminations, disbonds, and inclusions/porosity - and incorporate structural configurations of Boeing, Douglas, Airbus, and Fokker aircraft. They should also mimic porosity levels of repair processes that can mask NDI signals. These standards have been evaluated by a wide variety of NDI systems to determine the applications and limitations of each. The main issues involved in this activity were:

1) accounting for standards which are currently available,
2) developing a focus so that it results in a workable number of specimens; currently, the recognized number of variables makes the potential number of standards very large,
3) ultimately, producing reference standards which accommodate initial damage assessment as well as inspection of composite repairs,
4) applying current, as well as emerging, inspection technologies.

This activity was able to produce a workable number of reference specimens. Currently, the number of composite honeycomb construction scenarios makes the resulting number of standards very large and unmanageable. Inspection characterizations and equipment responses were used to determine the important variables needed in a composite reference standard thus eliminating unnecessary standard configurations.

The results from the honeycomb reference standard effort are formally documented in the SAE Aerospace Recommended Practice (ARP) 5606. The purpose of this ARP is to describe the design and production of composite honeycomb calibration standards to be used in ultrasonic, resonant, and tap test NDI equipment calibration for accomplishment of damage assessment and post-repair inspection of aircraft composites. These standards may also be appropriate for other NDT methods but will need to be assessed as appropriate prior to their use. The standards are representative of structures found on aircraft and include typical flaw scenarios such as disbonds and delaminations. These standards have been adopted by aircraft Original Equipment Manufacturers within procedures contained in their Nondestructive Testing Manuals. Depending on the nature of the inspection, it may be necessary to compensate for variations in material properties through the use of correction factors or by adjusting for these differences on the part or structure being inspected. In certain instances, it may be desirable or necessary to design a new reference standard to accommodate a specific inspection application. However, it is believed that these standards will accommodate NDT equipment set-up for most honeycomb structures found on commercial aircraft.

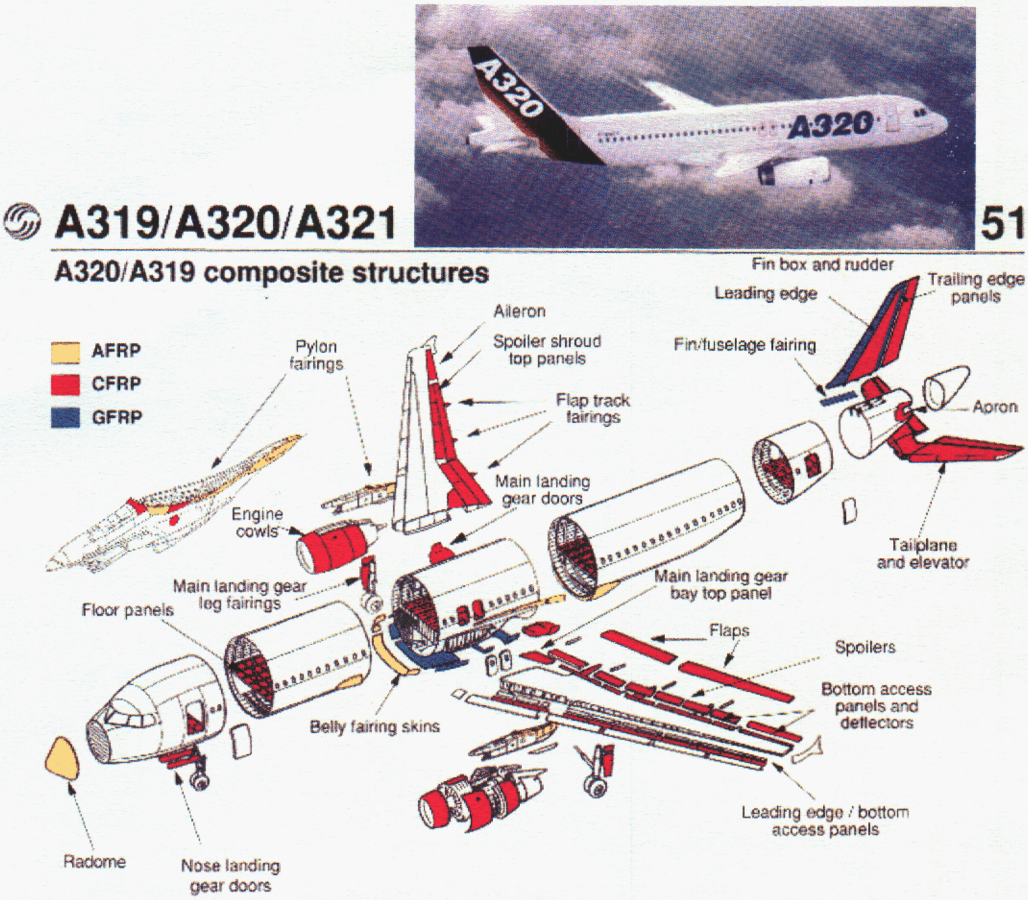

Figure 2: Use of Composite Structures on Airbus 320 Series Aircraft

1.3 COMPOSITE HONEYCOMB INSPECTION METHODS

High Frequency Bond Testing (HFBT) is often referred to as resonance testing. It is similar in application to contact ultrasonics in that a transducer with a hard wear face is acoustically coupled to the item under inspection using a liquid couplant. HFBT utilizes special narrow bandwidth transducers, which, when coupled to the item under test, produce a continuous sound field in the material. The test material, in turn, provides a mass loading on the transducer increasing the transducer bandwidth as well as changing the transducer's resonant frequency. Anomalies (such as disbonds) or changes in material thickness result in changes to the transducer loading that cause changes in transducer resonance. These changes are subsequently detected as differences in phase and amplitude of the electronic detection circuits. Acoustic impedance changes can be thought of as variation in the ability to transmit sound between the probe and the material under test. Changes in the materials acoustic impedance cause a corresponding change in the electrical impedance of the transducer and it is these electrical impedance changes that are monitored by the instrument. High Frequency Bond Testing has proved to be effective for inspection multilayer metal and non-metal laminates for the detection of disbonds as well as multi-ply, non-metallic composite structure for the detection of inter-ply delaminations.

Low Frequency Bond Testing (LFBT) refers to bondtesters that operate below 100 KHz and are generally called sonic bondtesters. They generally do not require the use of liquid couplant (dry coupled) and operate in the audio or near-audio frequency range. Different techniques for transmitting and receiving energy have been developed for low frequency bondtest applications. Each technique introduces a pressure wave into the specimen and then detects the transmitted or reflect wave. The pitch-catch impulse test method uses a dual-element, point contact, non-couplant, low frequency sonic probe. One element transmits acoustic waves into the test part and a separate element receives the sound. The sound propagates in a complex wave mode across the test piece between the probe tips. The return signals are processed and the difference between the effects of good and bad areas of the part along the sound path are analyzed and compared. A complex wave front is generated internally in the material as a result of velocity characteristic, acoustical impedance, and thickness. The time and amount of received energy is affected by the changes in material properties, such as thickness, unbonds and discontinuities. The instrument processes the received impulse and displays the received information on a phase and amplitude meter.

Mechanical Impedance Analysis (MIA) is the method of bond testing that compares the stiffness of a structure in contact with the probe tip. The stiffness of the bonded structure is a function of thickness, geometry, elastic variables and densities of the bonded components. The bonded structure under test is vibrated. Disbonds or other anomalies normally cause a reduction in mechanical impedance (stiffness) and can result in a phase or amplitude change to the displayed signal, depending on the frequency of the probe. The probe consists of two piezoelectric crystals with a driver positioned behind the receiver within the same holder. The driver converts electrical energy into sonic vibrations and the receiver, in direct contact with the test surface, converts the modified vibrations into electrical signals for processing by the instrument. If the probe is placed on an infinitely stiff structure and the driver crystal is set to vibrate at a given frequency, then the receiver crystal will compress and expand in opposition to the driver crystal (180° phase shift) at maximum signal amplitude. If the probe is now placed on an infinitely flexible structure (free air) and the driver set to vibrate at a given frequency, then the receiver crystal will simply move back and forth in space but will not be compressed or expanded and thus produce no output. Somewhere between these two extremes lies reality and in general a defect will produce a signal containing amplitude proportional to its stiffness with a possible phase change. The displayed information can be impedance plane (flying spot), meter deflection, or horizontal bar graph. Alarm thresholds can be used to provide audible or visual warnings.

Tap Testing is a manual method wherein a small diameter rod or hammer with a spherical tip is used to tap the surface of a structure while the human ear is used to monitor the audible results. Subtle variations in the audible response from the structure are detected by the inspector and used to infer the presence of flaws. The audible sound resonating from the part will be characteristic of the mass, cohesive stiffness, and the cross-sectional thickness of the part or assembly. The characteristics of the impact are dependent on the local impedance of the structure and on the mass of the tapper used. When a defective area is tapped, the higher structural vibration modes are not excited as strongly as when a structurally sound area is tapped. The sound produced from a defective area has less high frequency content and the structure sounds "duller." Electronic tap test instruments have been developed to automate the inspection process. Some of these instruments measure the duration of the impact while others measure the frequency content of the tap signal. Tap testing is most effective on honeycomb structure with thin face sheets.

Radiographic Inspection is performed by transmitting an X-ray beam through the part and onto film. The unabsorbed radiation exposes the film emulsion and the resultant image is called a radiograph. The radiograph is an orthographic projection or essentially a shadow picture of the part. Variations in density, thickness and composition of the part being inspected cause variations in the density of the developed image. A change in density can be caused by a change of part thickness, cracks, porosity, crushed honeycomb core or variation in the part composition such as the presence of fluid in a honeycomb or foreign material inclusion.

Thermography is a nondestructive inspection method that uses thermal gradients to analyze the physical characteristics of a structure such as internal defects. This is done by converting a thermal gradient into a visible image by using a thermally sensitive detector such as an infrared (IR) camera. By the judicious application of external heat sources, common aircraft defects can be detected by an appropriate infrared survey. The heat source, such as flash lamps, is used to raise the surface temperature of the structure. The subsequent heat transfer into the material is affected by any defects that may be present. The resulting temperature distribution is then recorded by the IR camera and displayed on the computer monitor.

Shearography – This is a wide area interferometric imaging technique that is capable of detecting micron-sized displacements of the surface of a structure. Shearography equipment monitors the surface of a structure for any changes in the surface strain field. Stressing the material in the appropriate way ensures that the subsurface anomalies are manifested on the surface of the structure. Shearography is implemented by comparing two interference patterns on a detector plane, typically "before" and "after" an object motion. If the motion, and subsequent out-of-plane deformations, cause changes in the optical path, then the speckle patterns differ. These images can be compared by subtraction or other algorithms to obtain an image of the object with fringe patterns superimposed. These fringe patterns can then be used to identify the presence, size, and depth of flaws in a structure.

2. PROJECT DESCRIPTION

This project addressed the need to introduce additional NDI for composite structures and established the sensitivities and limitations of potential NDI methods. One phase of this effort continued to develop composite reference standards and assessed improvements in composite inspections through their use. The reference standards include typical composite flaw scenarios - delaminations, disbonds, and inclusions/porosity - and incorporate structural configurations of Boeing, Airbus, Douglas, Embraer, Bombardier, and Fokker aircraft.

The basic tasks necessary to support this effort are as follows: 1) review composite structure designs of each OEM and discuss unique reference standard needs with the OEMs, 2) develop a series of processes for producing the various engineered flaws in the specimens, 3) apply NDI techniques and assess their applications and limitations, and 4) produce new or enhance existing composite NDI procedures through the use of the reference standards and possible application of improved NDT equipment.

A series of coupon specimens were produced in order to determine optimum ways of producing the disbonds, delaminations, and inclusions in the composite structures. Items varied included core thickness, core weight, skin material, cell size, skin thickness, and core material. NDI was applied to the specimens in order to assess the difficulties presented by the engineered flaws. The inspection results were used to identify the important variables which should be included in composite honeycomb reference standards. In this manner, the effects of each variable on NDI were assessed in order to provide justification for minimizing the number of calibration standards.

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3. DETERMINING KEY FACTORS AFFECTING INSPECTION

3.1 DEVELOPING APPROPRIATE RANGE OF HONEYCOMB SPECIMENS

A set of 64 honeycomb specimens were fabricated to isolate the effects of the construction variables (materials and flaw type) and bounding conditions on NDI. Table 1 summarizes these variables.

Table 1: Range of Material Variables Used to Bound Honeycomb Construction

Variable	Bounding Conditions	
	Minimum	**Maximum**
Laminate Material	Fiberglass	Carbon
Laminate Thickness	3 plies	12 plies
Honeycomb Core Material	Nomex	Fiberglass
Honeycomb Core Thickness	0.25"	2.0"
Honeycomb Cell Size	0.125"	0.25"
Honeycomb Core Density	2 lb./ft^3	8 lb./ft^3
Disbond and Delamination Flaws	At various depths	At various depths

The bounding conditions on each parameter represent the extreme values found in aircraft construction. The goal of this approach was to allow the results from this program to be applied to aircraft from all manufacturers. Figure 3 shows the design of the composite honeycomb panels used in this parametric study. Sixteen panels contained four different construction types (four quadrants) and isolated the effects of each of the variables listed above (2 extremes, 6 variables creates $2^6 = 64$ different specimens). NDI was applied to the specimens in order to assess the difficulties presented by the engineered flaws. The inspection results were used to identify the important variables which should be included in composite honeycomb reference standards. In this manner, the effect of each variable on NDI was assessed.

3.2 APPLICATION OF NDT EQUIPMENT

Multiple NDI techniques were applied to the 64 sandwich construction test specimens defined by the variable options. Upper and lower bounds were intentionally used for each construction variable in order to demonstrate which variable extremes have little or no effect on NDI. Common NDI responses at both ends of the variable extremes provided the engineering justification for minimizing the number of necessary reference standards.

The NDI techniques and specific equipment that were applied to the matrix of honeycomb test specimens were: low/high frequency bond testers (S-9 Sondicator, Bondmaster, and MAUS in resonance mode), through-transmission and pulse-echo (PE) ultrasonics (Staveley 136, Quantum, MAUS in PE mode), tap test (Mitsui Woodpecker, Digital Tap Hammer), thermography (Thermal Wave Imaging), and mechanical impedance analysis (MIA-3000, V-95 Bondcheck). Figure 4 shows inspection data being accumulated from different NDT equipment applications on the honeycomb panels.

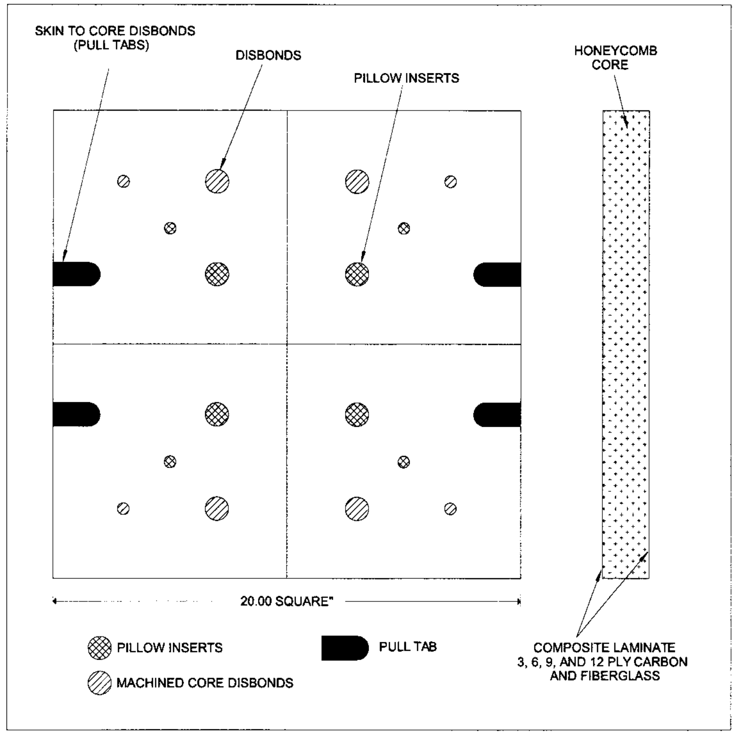


Figure 3: Design Drawing of Composite Honeycomb Panel Containing Four Different Construction Types and Engineered Flaws

3.3 USE OF SIGNAL-TO-NOISE VALUES TO IDENTIFY KEY NDI VARIABLES

In order to intercompare the results from different NDI methods that use different indicators to infer the presence of defects, each inspection measured the signal-to-noise ratio (S/N) of each defect vs. the surrounding good structure. The noise level was determined by examining the output variation corresponding to inspections along adjacent sections of good structure. This was compared to the signal obtained during inspections of the flawed areas.

BS = base signal; peak signal at unflawed area
NS = noise signal; (max-min)/2 over range of unflawed area in each quadrant
FS = flaw signal; peak signal at each flaw site
S/N = signal-to-noise ratio

$$S/N = \frac{FS - BS}{NS} \tag{1}$$

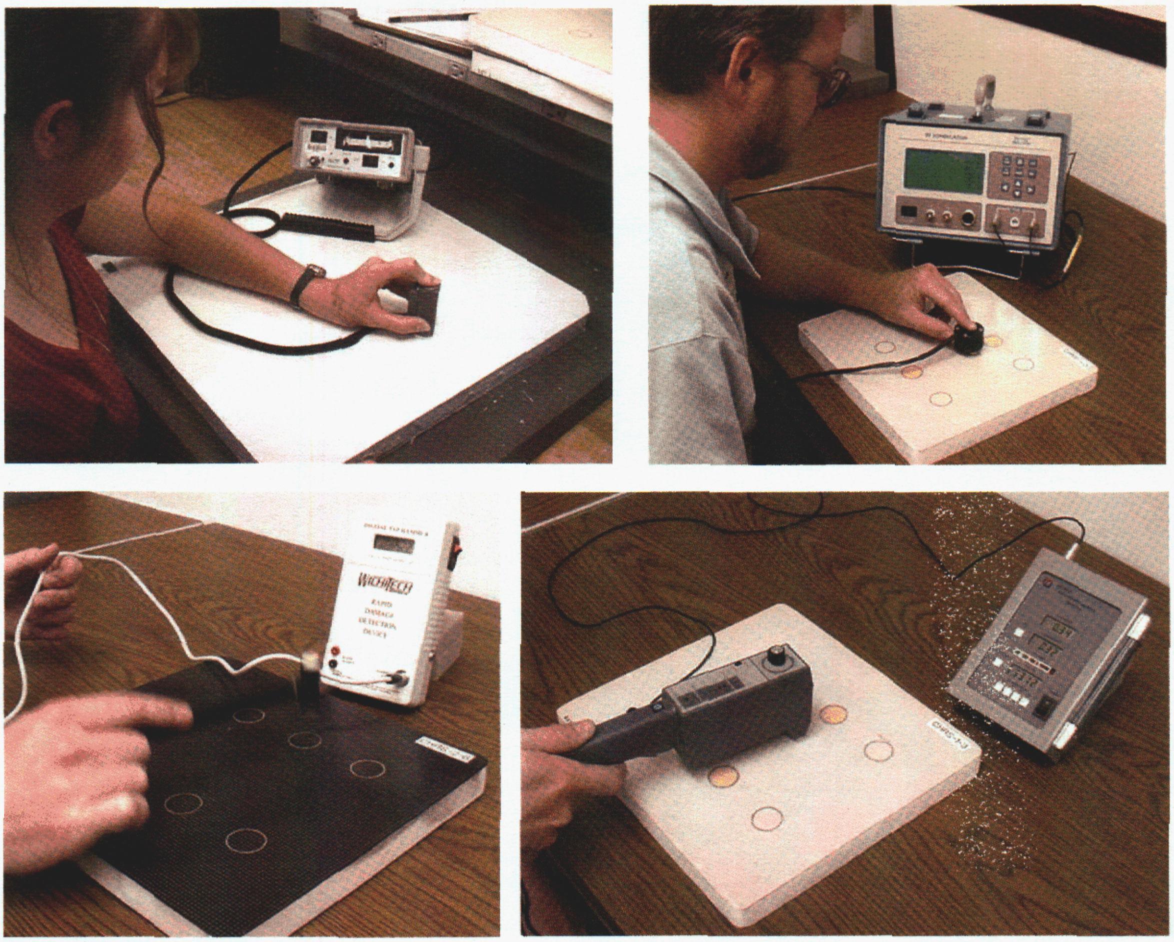
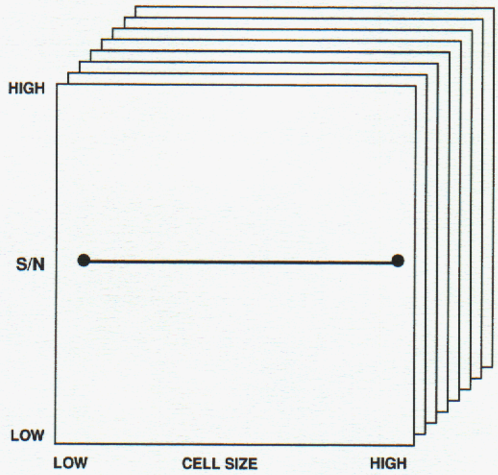


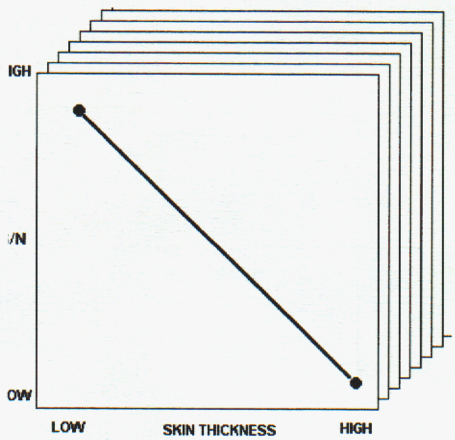
Figure 4: Application of Inspection Equipment on Array of Honeycomb Panels (clockwise from upper left) - V-95 Bondcheck, S-9 Sondicator, Woodpecker, Digital Tap Hammer

Testing using this scheme did not require calibration on a "median" or "neutral" reference standard. The key measurement for each case was the difference between unflawed areas of the test panel and the defect area. Hypothetical signal-to-noise testing results for different variable effects are shown in Figure 5A and 5B. If a signal-to-noise value remains constant over the full range of honeycomb cell sizes (see Figure 5A), then it can be inferred that increasing cell size has no effect on defect detectability. Therefore a reference standard with any cell size can be used to inspect structure with cell sizes of 1/8" to 1/4". However, if the signal-to-noise ratio changes significantly as panels of different skin thickness are inspected (see Figure 5B), then skin thickness is an important factor in setting up for honeycomb inspections. Therefore the reference standards must have skins that closely represent the structure to be inspected (small step increments).



Effect of Cell Size
(8 replicates)

Figure 5A: Unchanged Signal-to-Noise Ratio Indicates That Increasing Cell Size Has No Effect on Defect Detectability



Effect of Skin Thickness
(8 replicates)

[Result: reference standards must have skin thicknesses that closely represent the structure to be inspected]

Figure 5B: Changing Signal-to-Noise Ratio Indicates That Increasing Skin Thickness Has A Major Effect on Defect Detectability

3.4 NDI DATA ANALYSIS

The inspection results were used to identify the important variables which should be included in composite honeycomb reference standards. The raw X-Y and C-scan data, reduced into signal-to-noise data using equation (1), was analyzed using a variance analysis. The statistical analysis of the data was conducted in order to place the effects of flaw and construction variables into "major," "minor," and "minimal" categories. The analysis determined the effect of variables alone (e.g. impact of material thickness) and in two and three variable combinations (e.g. impact of core type in combination with laminate type). The flaw types analyzed were: 1" pull tab (delaminations), 1" milled core (disbond), and 1" pillow inserts (delaminations). The six construction factors included in the analysis were: laminate type (carbon or fiberglass), laminate thickness (3 plies or 12 plies), honeycomb type (fiberglass or Nomex), honeycomb thickness (0.25" or 2"), honeycomb cell size (0.125" or 0.25"), and honeycomb density (2 lb./ft.^{3} or 5 lb./ft.^{3}). Sample results from NDT equipment are shown in Figure 6. In this figure, the scatter of signal responses from unflawed areas is used to arrive at the noise level. This is then compared to the magnitude of the signals obtained from the test specimen's flawed regions. Data from equipment providing digital signal values (e.g. Woodpecker, S-9 Sondicator) were logged directly into a data acquisition computer in real time.

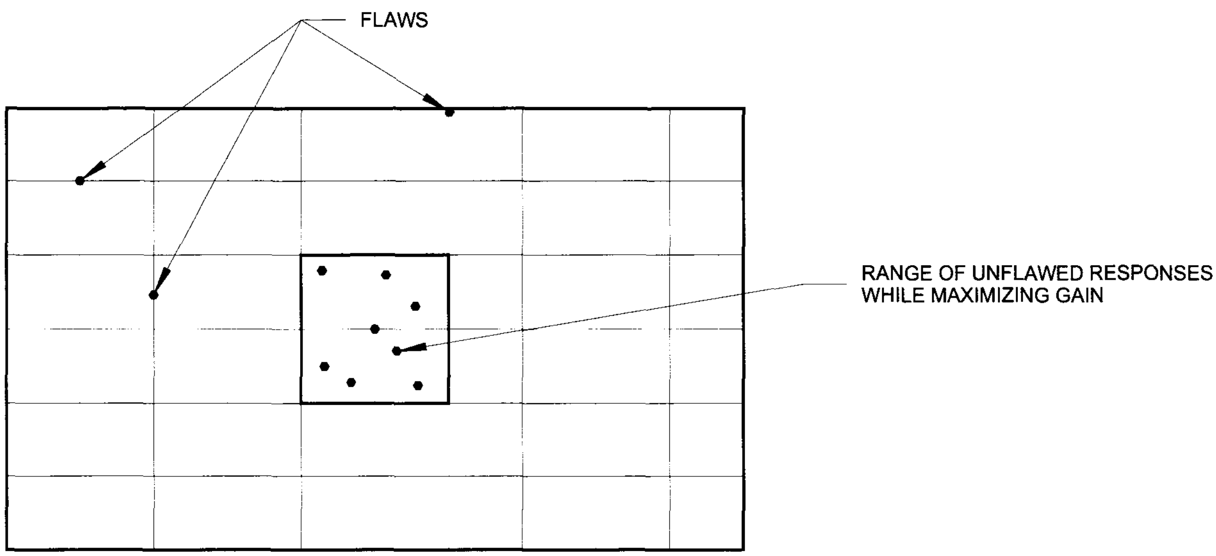


Figure 6: Validation of Honeycomb Standards -Sample Hand-Held NDT Results from Bondmaster Device

3.5 RESULTS FROM VARIANCE ANALYSIS

The six factors defining typical honeycomb construction on aircraft were incorporated into the construction of test structures. The factors included were: laminate type, laminate thickness, honeycomb type, honeycomb thickness, honeycomb cell size, and honeycomb cell density. Two levels for each factor yields a total of 64 (=2^6) test structures. In each of these test specimens typical flaw types were included. For a given NDT instrument, the signal–to–noise (S/N) response was taken at each of the flaws in each of the 64 test specimens. From this data a regression–like model was estimated in which the S/N response is expressed as a function of the levels of each of the six factors and the two–way combinations of the factors. The residual variation is used to gage whether the individual factors, or a combination of factors, explain more of the variation than can be attributed to chance. If a factor or combination of factors does not explain a significant amount of the variation, then it is removed from the model. A hierarchical structure of the model is maintained in that a combination of factors is not retained without keeping the lower order combinations of the same factors in the model

Figure 7 illustrates conditions in which the laminate type (LTY) and laminate thickness (LTH) display an interaction effect. In this example it is seen that the mean response for carbon changes little in going from 3 plies thick to 12 plies thick. However, the mean response rises appreciably in going from 3–ply to 12–ply thick specimens when the material is fiberglass. In both cases the amount of change is judged with respect to the error bars that are shown. The error bars are determined from the residual variation after fitting the model and show that the rise in response is significant. The resultant regression model therefore, contains factors of LTY, LTH, and the product of the two variables (LTY * LTH).

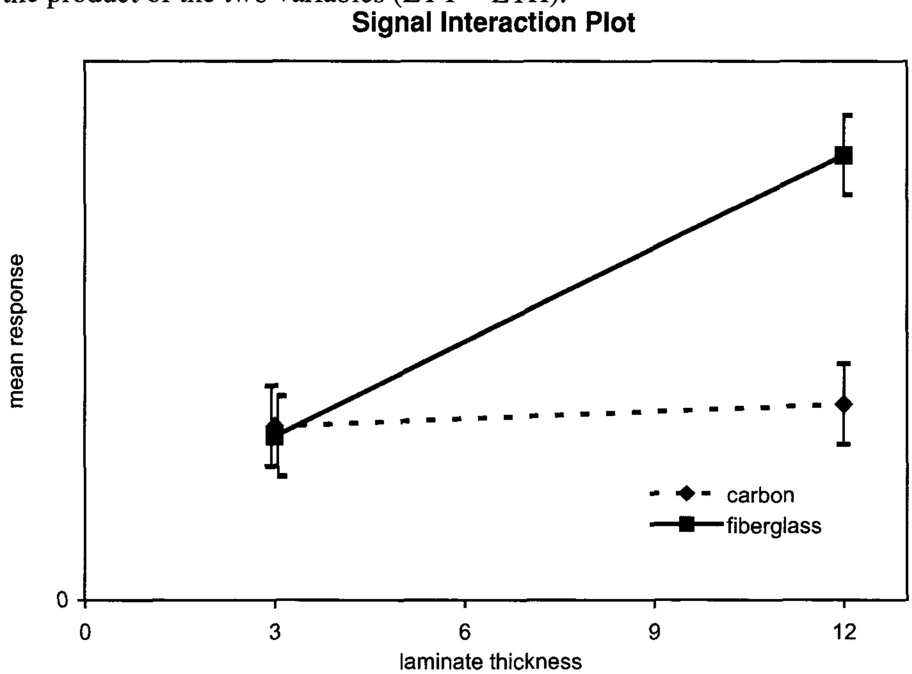


Figure 7: Signal Interaction Plot Depicting Use of Signal-to-Noise Ratio Data in Variance Analysis

The model outlined above was fitted independently for various NDT techniques and for various types of flaws. The factors that consistently show up as being significant for the various techniques constitute factors that need to be reflected in composite reference standards. If this includes an interaction, such as that shown in the example, then those combinations of the factors need to be reflected in the standards as well.

The statistical analysis of the round-robin test series produced the following conclusions:

a. For the pillow insert (delamination) flaws, the dominant effects across all inspections were laminate thickness, laminate type, and honeycomb type.
b. For the milled core (disbond) flaws, the dominant effects across all inspections were laminate thickness, honeycomb thickness, and honeycomb type.
c. For the pull tab (delamination) flaws, the dominant effects across all inspections were laminate thickness, laminate type, and honeycomb type.

This data indicates that composite honeycomb reference standards should consider the following variable ranges: laminate thickness (3 ply to 12 ply), laminate type (both fiberglass and carbon), honeycomb type (fiberglass and Nomex), and honeycomb thickness (1/4" to 2"). After additional data analysis, the primary variables were determined to be laminate type and thickness and honeycomb type.

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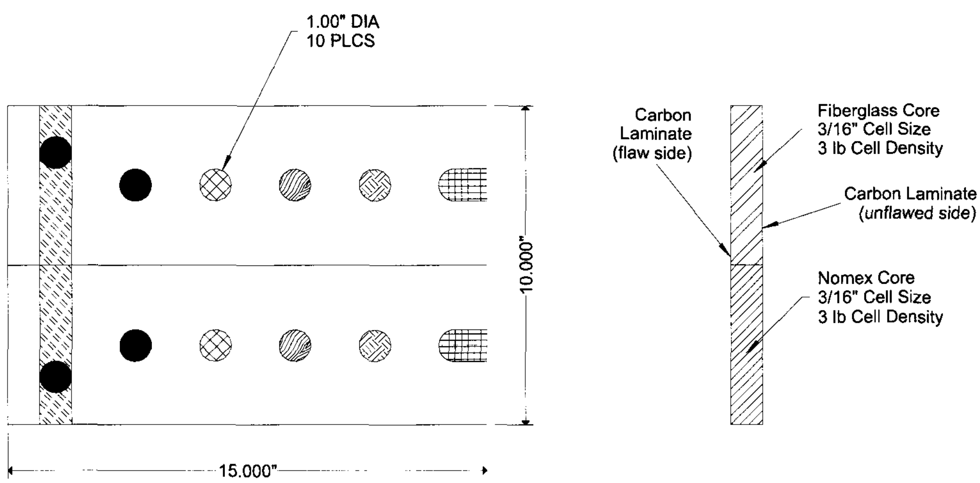
4. VALIDATION OF MINIMUM HONEYCOMB REFERENCE STANDARD SET

4.1 PROTOTYPE MINIMUM REFERENCE STANDARD SET

The results presented above led to the production of a prototype minimum reference standard set that included the important variables for the successful inspection of composite honeycomb structure: laminate thickness, laminate type, and honeycomb type. The construction characteristics of the prototype honeycomb set are summarized in Table 2. Disbonds and delaminations were placed together in a single standard. Thus, there were eight standards manufactured: a 3, 6, 9, and 12 ply laminate with carbon or fiberglass skins and each containing both Nomex and fiberglass cores. Figure 8 shows the basic honeycomb design approach. Notice that these initial prototype standards contain two methods for creating interply delaminations (Teflon inserts and Pillow inserts) and two methods for creating skin-to-core disbonds (machined core and pull tabs). These methods will be described further in Section 5.0 along with the justification for final selection of a single, optimum method for engineering each type of flaw. Potted core and core splice regions are also included in the standards in order to aid the interpretation of NDI signals. These were included to help minimize false calls caused by the presence of potted cores or core splices in aircraft structure that will alter NDT equipment readings.



MACHINE CORE (depth Of 0.250" in honeycomb)

PILLOW INSERTS (located two plies up from core material)

TEFLON INSERTS (located two plies up from core material)

POTTED CORE (use Ciba-Geigy CG-1305 Epoxy)

CORE SPLICE (use AF-3028 potting compound)

PULL TAB (disbond located between laminate and core material)

Figure 8: Prototype Set of Honeycomb Reference Standards (same design for both carbon and fiberglass skin)

Table 2: Honeycomb Reference Standards Used to Set Up NDI Equipment for Inspection Exercise

Variables Addressed in Prototype Composite Honeycomb Standard Set						
Flaw	**Laminate Type**	**Laminate Thickness**	**Honeycomb Type**	**Honeycomb Thickness**	**Cell Size**	**Cell Density**
Delam.	Carbon	3, 6, 9, 12 plies	Nomex	1"	3/16"	3 lb.
Disbond	Carbon	3, 6, 9, 12 plies	Nomex	1"	3/16"	3 lb.
Delam.	Fiberglass	3, 6, 9, 12 plies	Nomex	1"	3/16"	3 lb.
Disbond	Fiberglass	3, 6, 9, 12 plies	Nomex	1"	3/16"	3 lb.
Delam.	Carbon	3, 6, 9, 12 plies	Fiberglass	1"	3/16"	4 lb.
Disbond	Carbon	3, 6, 9, 12 plies	Fiberglass	1"	3/16"	4 lb.
Delam.	Fiberglass	3, 6, 9, 12 plies	Fiberglass	1"	3/16"	4 lb.
Disbond	Fiberglass	3, 6, 9, 12 plies	Fiberglass	1"	3/16"	4 lb.

** 3 lb. Density unavailable in fiberglass honeycomb*

4.2 VALIDATION TESTING APPROACH

A sequence of NDI tests were performed to determine that this prototype honeycomb reference standard set was able to support inspections over the full range of honeycomb construction scenarios. Appropriate OEM inspection procedures and manufacturer equipment calibration procedures were followed. An alarm threshold was set and flaws in the standard were assessed. After setting up the equipment on each flaw/skin thickness scenario, the set of 64 "aircraft" panels were inspected. Amplitude and phase data were used to assess the viability of the standards. If the full array of 64 specimens – which bound the composite honeycomb structure on aircraft - could be adequately inspected using the minimal standard set, then this study successfully identified the key variables and provided justification for excluding other honeycomb construction variables from the set.

4.2.1 Set of NDI Tests

The inspection scenario listed below was used to conduct the NDI experiments. By setting up the equipment on 6 ply laminates and then inspecting 3, 9, and 12 ply specimens it was also possible to assess whether or not exact laminate thickness matches are required (i.e. the allowable variation between laminate thickness used in set-up and laminate thickness in part being inspected while still obtaining a successful inspection).

Set 1: Inspect the set of 32 "aircraft" carbon skin honeycomb specimens after setting equipment up on:

12 ply, carbon laminate, Nomex honeycomb, disbond flaws
12 ply, carbon laminate, Nomex honeycomb, delamination flaws
12 ply, carbon laminate, fiberglass honeycomb, disbond flaws
12 ply, carbon laminate, fiberglass honeycomb, delamination flaws

9 ply, carbon laminate, Nomex honeycomb, disbond flaws
9 ply, carbon laminate, Nomex honeycomb, delamination flaws
9 ply, carbon laminate, fiberglass honeycomb, disbond flaws
9 ply, carbon laminate, fiberglass honeycomb, delamination flaws
6 ply, carbon laminate, Nomex honeycomb, disbond flaws
6 ply, carbon laminate, Nomex honeycomb, delamination flaws
6 ply, carbon laminate, fiberglass honeycomb, disbond flaws
6 ply, carbon laminate, fiberglass honeycomb, delamination flaws
3 ply, carbon laminate, Nomex honeycomb, disbond flaws
3 ply, carbon laminate, Nomex honeycomb, delamination flaws
3 ply, carbon laminate, fiberglass honeycomb, disbond flaws
3 ply, carbon laminate, fiberglass honeycomb, delamination flaws

Set 2: Inspect the set of 32 "aircraft" fiberglass skin honeycomb specimens after setting equipment up on:

12 ply, fiberglass laminate, Nomex honeycomb, disbond flaws
12 ply, fiberglass laminate, Nomex honeycomb, delamination flaws
12 ply, fiberglass laminate, fiberglass honeycomb, disbond flaws
12 ply, fiberglass laminate, fiberglass honeycomb, delamination flaws
9 ply, fiberglass laminate, Nomex honeycomb, disbond flaws
9 ply, fiberglass laminate, Nomex honeycomb, delamination flaws
9 ply, fiberglass laminate, fiberglass honeycomb, disbond flaws
9 ply, fiberglass laminate, fiberglass honeycomb, delamination flaws
6 ply, fiberglass laminate, Nomex honeycomb, disbond flaws
6 ply, fiberglass laminate, Nomex honeycomb, delamination flaws
6 ply, fiberglass laminate, fiberglass honeycomb, disbond flaws
6 ply, fiberglass laminate, fiberglass honeycomb, delamination flaws
3 ply, fiberglass laminate, Nomex honeycomb, disbond flaws
3 ply, fiberglass laminate, Nomex honeycomb, delamination flaws
3 ply, fiberglass laminate, fiberglass honeycomb, disbond flaws
3 ply, fiberglass laminate, fiberglass honeycomb, delamination flaws

In addition to checking for flaw detection, sensitivity was also studied through the acquisition of signal-to-noise (S/N) information as in the first round of inspections. Flaw detection required at least a 2:1 ratio of signal-to-noise. S/N measurements were also made on the standards. The equipment, probe (frequency), and gain were optimized on the standard. After this calibration was complete, equipment null adjustments were allowed during the course of the panel inspections but there were no adjustments in gain allowed. The equipment was set up in accordance with existing NDI procedures or the operator's manual for the NDT equipment. The set of inspections listed above resulted in 32 specimens X 16 configurations = 512 inspections per set for a total of 1024 specimen inspections (fiberglass and carbon skin sets).

4.2.2 Equipment to Be Applied

It was decided that the best equipment to be used in this test series was the S-9 Sondicator. None of the other equipment listed in Section 3.2 allowed adjustments to be made (other than null point) when moving from the standard to the aircraft. Thus, they did not need to be revisited for this exercise. Inspections were performed using the S-9 in both manually deployed mode (performed by AANC) and automated, C-scan mode (performed by Boeing).

4.3 VALIDATION TESTING RESULTS

Signal-to-noise (S/N) results from the panels were all greater than 2:1 and indicated acceptable flaw detection over the entire range of honeycomb types. Thus, the set of eight prototype honeycomb reference standards described above are able to support the inspection of honeycomb aircraft structure. Furthermore, after setting up the NDT instrument on a 6 ply standard, it was possible to inspect 3 and 9 ply aircraft panels. However, the flaw sensitivity was not as good as when closer ply matches were used for calibration. As a result, the prototype standard set was not altered and it was concluded that 3, 6, 9, and 12 plies are needed to set up NDT equipment for the expected range of laminate skin thicknesses. Finally, NDI testing using bond testers (high and low frequency), pulse-echo ultrasonics, and mechanical impedance analysis demonstrated the difficulty of inspecting structures with 12 or more plies. While acceptable S/N results could often be obtained, the inspection results were not consistent. A comprehensive experiment is being conducted by our CACRC Inspection Task Group team that will quantify the ability of these conventional inspection methods, along with applicable advanced NDI methods, to locate and size flaws in composite honeycomb aircraft structures [3-5].

5. ASSESSMENT OF FLAW ENGINEERING METHODS TO OPTIMIZE STANDARD DESIGN AND FABRICATION

5.1 REFERENCE STANDARD DESIGN AND FABRICATION

Additional field testing was performed to complete the validation of the prototype honeycomb reference standard set (see Section 6.0 “Final Airline and OEM Validation”). However, before proceeding with this final phase of the validation, it was decided to reach some conclusions on the standard fabrication process. Several of the NDI tests highlighted some inconsistencies in the flaw manufacturing methods. Pillow insert flaws were used because it was thought that they could provide realistic flaw responses. However, it was determined that the response from the disbonds engineered with pillow inserts sometimes did not provide a sufficient deviation from the noise floor to allow for clear flaw detection. Inspection results from the entire suite of specimens generated thus far in the study proved that machining the honeycomb core (recessing) away from the laminate provides the best way of producing reliable skin-to-core disbond flaws. This method also produces flaw sites that can support tap testing. The remaining question was how to realistically and repeatably produce interply delamination flaws.

5.2 CHARACTERIZATION RESPONSE FROM TRIAL SPECIMENS AND PROTOTYPE STANDARDS

To answer this question, a series of inspections were carried out on trial standards that were manufactured with various candidate methods for engineering delamination flaws. Figure 9 shows the engineering drawing for one of these honeycomb specimens for evaluating flaw insertion methods. One carbon and one fiberglass skin specimen was produced with this flaw layout. The three methods employed to engineer the delamination flaws were: 1) pillow insert consisting of Kapton tape around 4 layers of tissue paper, 2) brass shims coated with a Silicon mold release to prevent bonding to the plies, and 3) Teflon disk inserts. Each flaw method was used to generate three like delamination flaws in order to test for repeatability, as well as to statistically determine the amount of NDI signal disruption generated by the flaw method. Note also that the trial specimen includes potted core and core splice areas. In order to expand the utilization of these standards, potted core and core splice areas were included as a tool to aid the interpretation of NDI signals. This will help minimize false calls caused by the presence of potted cores or core splices that can alter NDT equipment readings.

5.3 METHODS FOR ENGINEERING FLAWS

a) Disbonds - Machining the honeycomb core (recessing) away from the laminate provides the best way of producing reliable skin-to-core disbond flaws. This method also produces flaw sites that can support tap testing. Pull tabs were also included as an alternate disbond engineering method, however, the required proximity of the pull tab flaws to the edge of the specimen produces boundary condition effects in some NDI methods and limits the value of these types of flaws.

b) Delaminations – The goal was to realistically and repeatably produce interply delamination flaws. The methods were repeated multiple times to assess consistency. The methods employed to engineer the delamination flaws were as follows: 1) pillow insert, 2) brass shims coated with a Silicon mold release agent, and 3) Teflon inserts (individual inserts with 0.003", 0.005", and 0.008" thickness and two plies of stacked 0.003" inserts). Figure 10 shows how the pillow inserts were fabricated. Experiments determined that the four layers of tissue paper are needed to produce a uniform and repeatable interruption (i.e. flaw indication) of an interrogating NDI signal.

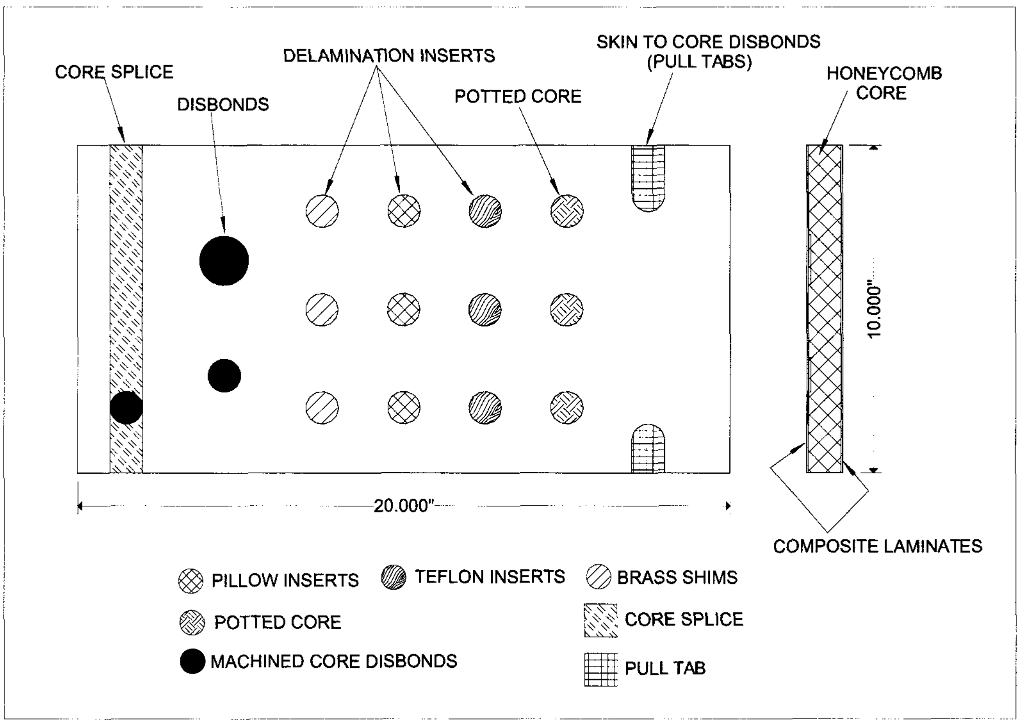


Figure 9: Engineering Drawing to Evaluate Honeycomb Reference Standard Design and Fabrication

c) Potted Core and Core Splice (use as reference not calibration) - The trial specimen included potted core (BMS-528 Type 7 material) and core splice (AF-3028 material) areas. In order to expand the utilization of these standards, potted core and core splice areas were included as a tool to aid the interpretation of NDI signals. A process was developed wherein the potting material was placed in a vacuum chamber just prior to insertion in the core cells with a syringe. This produced a uniform potting and no porosity was visible in subsequent TTU inspections. Figure 11 shows the process for creating a potted core area in the honeycomb. A profile of a machined core region to create disbonds is also shown in this schematic.

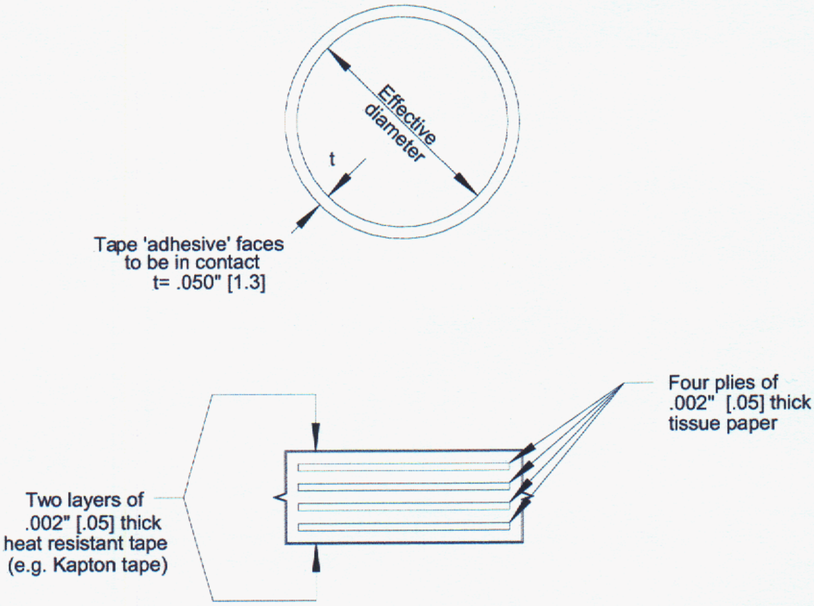


Figure 10: Construction of Pillow Insert for Delamination Flaws

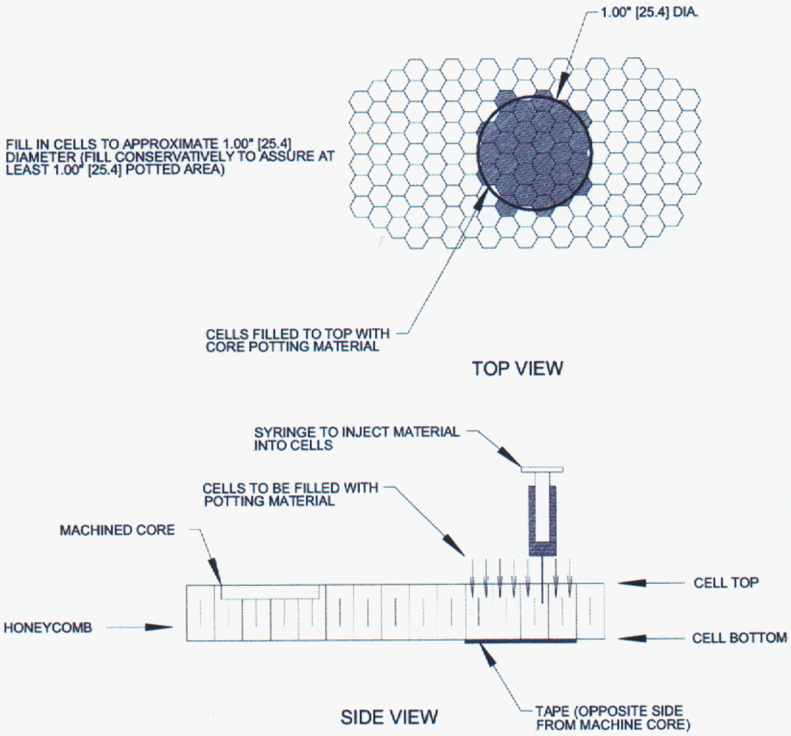
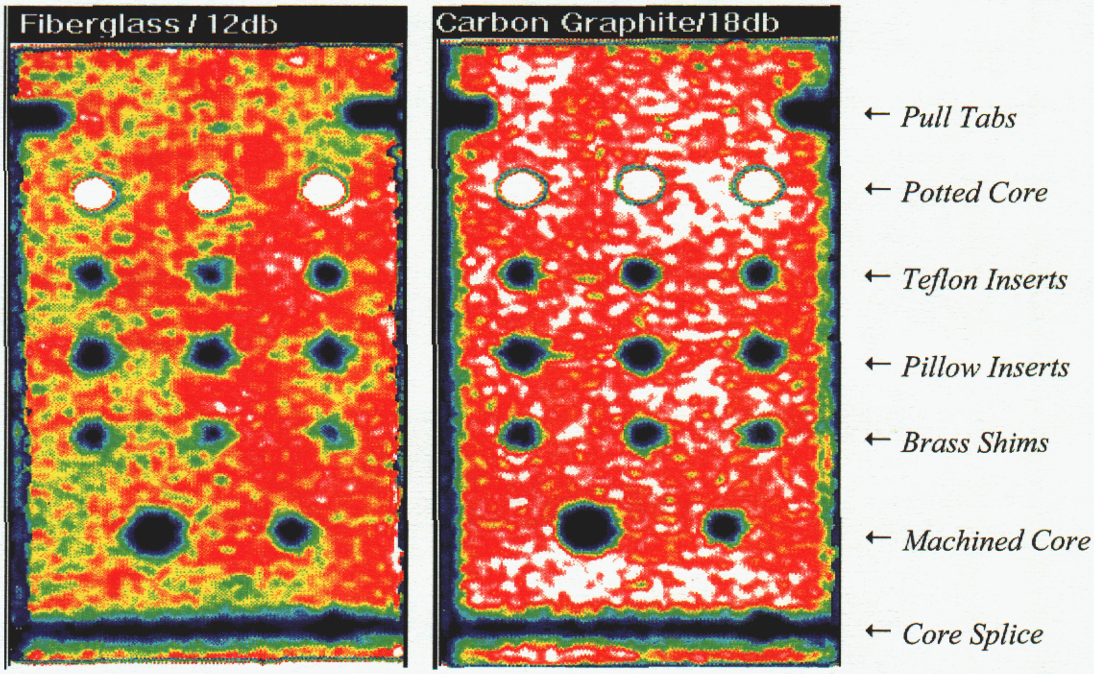


Figure 11: Process for Creating Potted Honeycomb Core Areas

Through-transmission ultrasonics (TTU) was used to evaluate the flaw engineering methods since it has the best resolution and can provide quantitative signal attenuation values. TTU results for the trial honeycomb panels (see Fig. 9 for design drawing) are shown in Figure 12. All of the flaws can be clearly seen, however, the Teflon inserts and the brass shim inserts, second and fourth row down respectively, are smaller than their original 1" diameter area. This is probably due to ingress of the adhesive around the perimeter of these inserts and an associated coupling of the ultrasonic signal. As a result, attenuation levels in the Teflon and brass shim areas are less than those measured in the pillow inserts regions. Furthermore the pillow insert regions were able to retain their 1" diameter size. The machined core disbond areas were also very consistent with uniform signal attenuation levels. Finally, the potted core regions shown in the top row provided enhanced signal coupling and an associated decrease in ultrasonic attenuation through the panel. This is expected and the results show a very uniform and repeatable signal level to support NDT calibration.



Fiberglass Skin **Carbon skin**

Figure 12: Through-Transmission Ultrasonic Inspection of Honeycomb Panel - Assessment of Methods to Engineer Repeatable Flaws

Two other laminate skin specimens were produced to complete the evaluation of various delamination inserts. TTU results from scanning these two skins are shown in Figures 13 and 14. This exercise determined that the use of multiple layers (i.e. 4 plies of tissue in pillow inserts and two plies of Teflon stacked together) produce better signal attenuation and more repeatable flaws than single ply inserts, even if the single plies are thicker. These results indicate that the best method(s) to engineer delaminations are: 1) pillow inserts with 4 plies of tissue, and 2) Teflon inserts with 2-3 plies of 0.003" material. Measured signal attenuation through these disbond flaws ranged from 26 dB to 46 dB. Bondmaster and S-9 inspections also support the use of these flaw insertion methods. A minimum of 18 dB attenuation is required at the flaw sites.

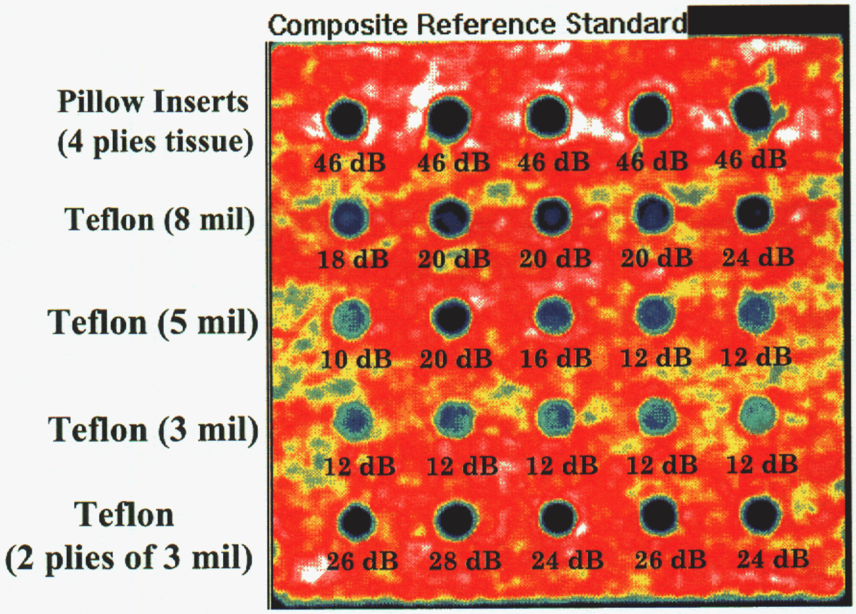


Figure 13: TTU C-Scan Showing Signal Attenuation Produced by Inserts; Tests on 6 Ply Fiberglass Laminate

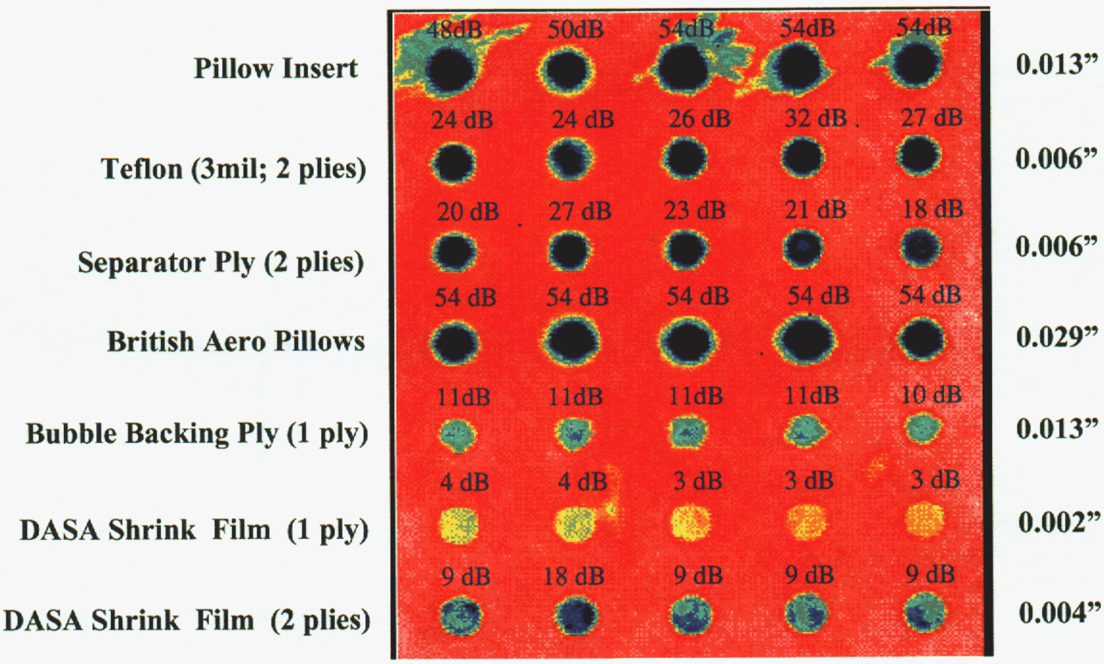


Figure 14: TTU C-Scan Showing Signal Attenuation Produced by Inserts; Tests on 6 Ply Fiberglass Laminate (Numbers in Right Hand Column Lists Thickness of Inserts)

5.4 FINAL DESIGN OF HONEYCOMB NDI REFERENCE STANDARD SET

The information accumulated from the tasks described above was used to arrive at the final design for the honeycomb reference standard set. An engineering drawing of the final honeycomb NDI reference standard design is shown in Figure 15. Figure 16 shows a cross section view highlighting the features of the standards. The design accommodates items identified as critical calibration elements such as edge distance around flaws and flaw spacing. The design allows for proper probe deployment on both good and flawed structure. Pillow inserts and machined core were selected as the best methods to produce interply delaminations and skin-to-core disbonds respectively. Potted core and core splice regions were also retained in order to aid the interpretation of NDI signals. This will help minimize false calls caused by the presence of potted cores or core splices in aircraft structure that can alter NDT equipment readings.

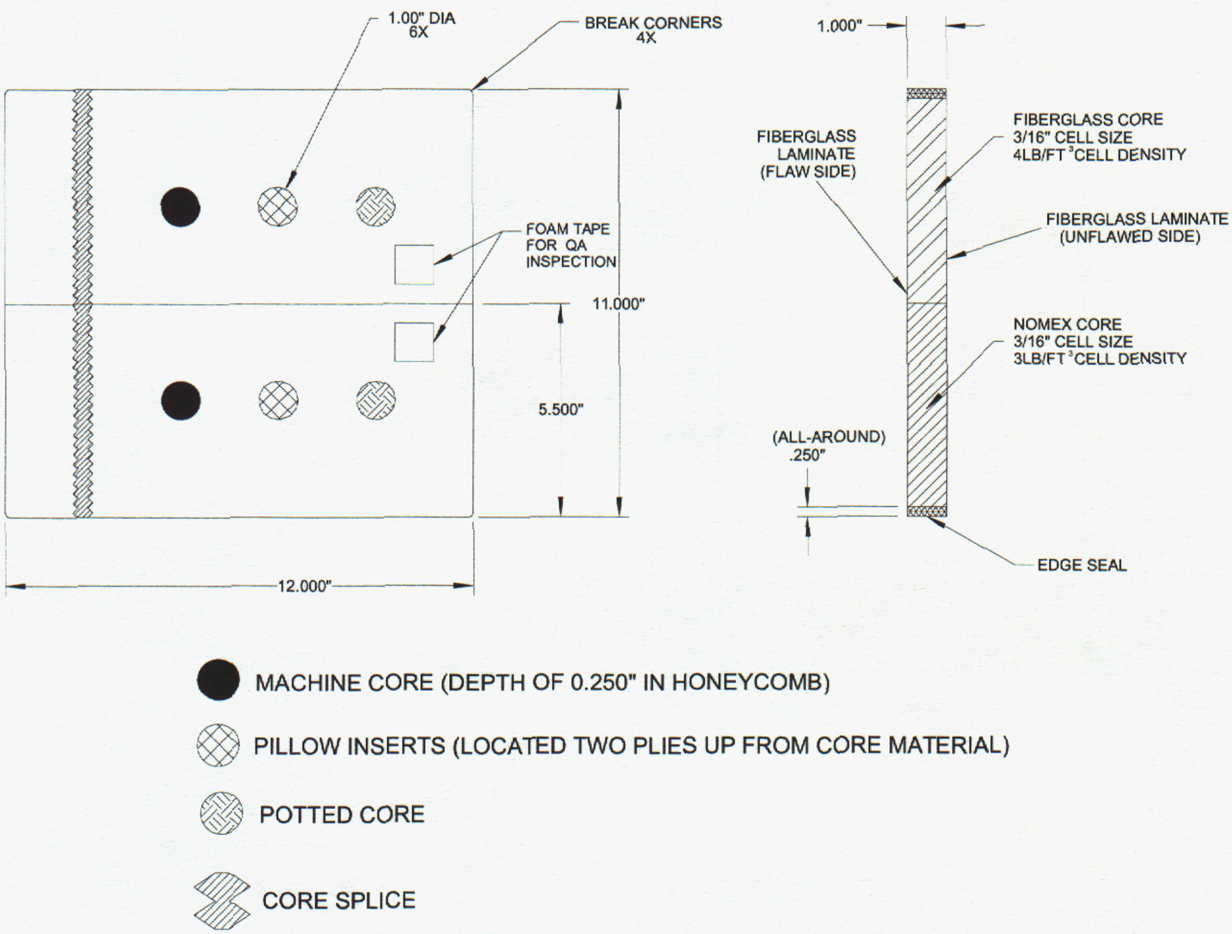


Figure 15: Final Design of Honeycomb NDI Reference Standards (Same Designs for Carbon & Fiberglass Skin and Repeated for 3, 6, 9, and 12 Plies)

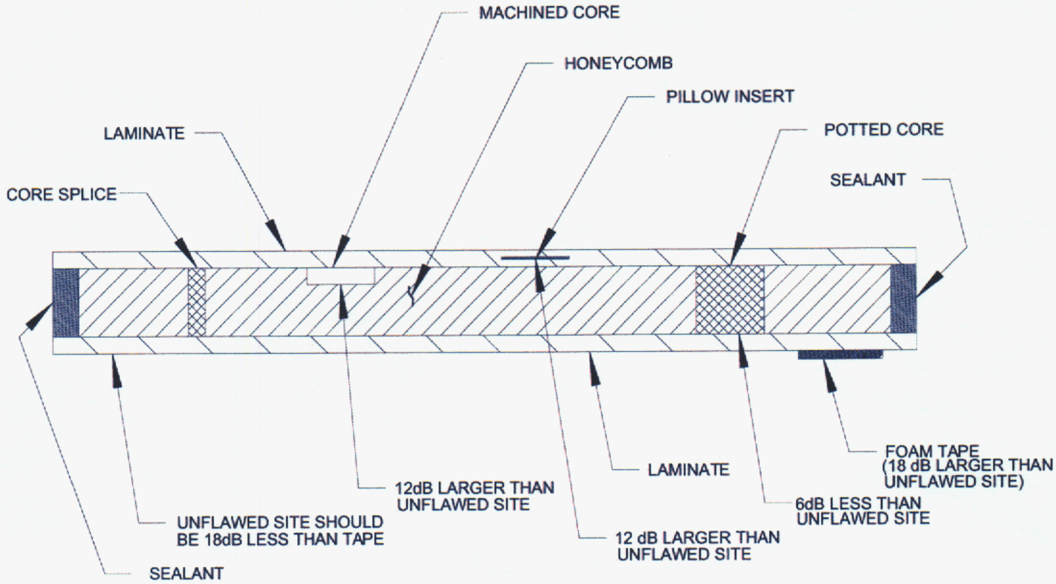


Figure 16: Cross Section of Final Honeycomb Reference Standard Design

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6. AIRLINE AND OEM ASSESSMENT

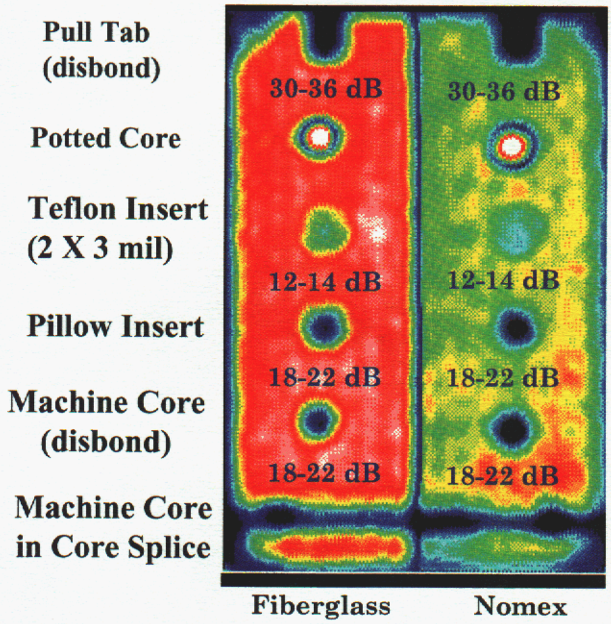
Another set of NDI tests were performed by inspectors and NDI engineers at various airline maintenance depots and aircraft manufacturer shops to further evaluate the honeycomb reference standard set. Participants included Boeing, Northwest Airlines, United Airlines, British Aerospace, and Airbus Industries. Testing followed the same approach as described in Section 4.0 "Validation of Minimum Honeycomb Reference Standard Set." The purpose of this testing was to allow manufacturers, who are approving the standards for use, and airlines, who will be the end users of the standards, to make in-house assessments on whether the standards can fully support inspections over the full range of honeycomb construction scenarios.

6.1 BOEING EVALUATIONS

Appropriate OEM inspection procedures and manufacturer equipment calibration procedures were followed. After setting up the equipment on each flaw and skin thickness/type scenario in the eight-panel reference standard set, the suite of 64 "aircraft" panels described in Section 3.0 were inspected. Amplitude and phase data were used to assess the viability of the standards. If the full array of 64 specimens – which bound the composite honeycomb structure on aircraft - could be adequately inspected using the minimal standard set, then the honeycomb NDI reference standards listed in Table 2 do indeed contain the key variables needed to support inspections of all honeycomb structures. These results also provide the justification for excluding other honeycomb construction variables from the set. Flaw detection was determined by whether or not sufficient signal-to-noise (S/N) levels were attained during the inspections. Flaw detection required at least a 2:1 ratio of signal-to-noise.

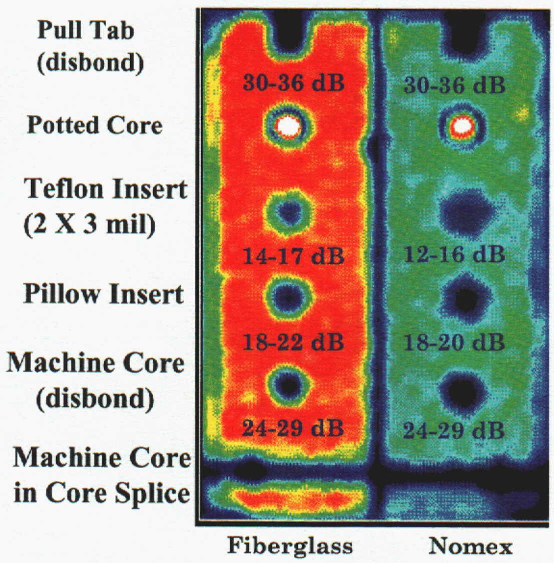
First, S/N measurements were made on the standards themselves. Figures 17 and 18 show TTU C-scan results from the prototype honeycomb standards (see Fig. 8 for schematic). A minimum of 18 dB attenuation is required at the flaw sites. It can be seen that the machined core (MC) disbonds and the pillow insert (PI) delaminations provide the necessary attenuation but the Teflon insert (TI) flaws are slightly below the acceptable level. This is the reason why the Teflon inserts were abandoned in favor of the pillow inserts for the final reference standard design (see Fig. 15).

Similar S/N results were obtained by Boeing during their standard assessment testing. The TTU inspection scans shown in Figures 19 and 20 possess the same characteristics in that MC and PI flaws are clearly identified while the TI are detected but do not show up as strongly. Boeing then inspected the suite of 64 "aircraft" panels using conventional NDT equipment. The aircraft panels were inspected following equipment set-up on the honeycomb reference standards. An extensive set of validation testing data was accumulated from both hand-held and C-scan NDT devices. Flaw detection results, based on signal-to-noise thresholds, indicate acceptable flaw detection over the range of honeycomb types found on aircraft. It was concluded that this study produced a good honeycomb reference standard set. Also, it was demonstrated that the prototype honeycomb standards are able to support a secondary goal of the project: establish limitations of NDI techniques. It was observed that the delaminations were optimally found by a HFBT method while the disbonds were more accurately identified by LFBT methods. It was also found that, unless the inspection is looking for large flaws in excess of 2" in diameter, tap testing is limited to structures containing 9 plies or less. Prototype honeycomb reference standard sets were used by United Airlines and Northwest Airlines to study how they function in the field. They were successfully evaluated on damaged honeycomb structure removed from aircraft and on honeycomb structure currently on aircraft.



Note: Attenuation numbers are artificially low due to size of transducer and signal transmission around edges

Figure 17: Sandia Labs (AANC) TTU Scan of 9 Ply <u>Fiberglass</u> Skin Over 1" Fiberglass and Nomex Honeycomb



Note: Attenuation numbers are artificially low due to size of transducer and signal transmission around edges

Figure 18: Sandia Labs (AANC) TTU Scan of 9 Ply <u>Carbon</u> Skin Over 1" Fiberglass and Nomex Honeycomb

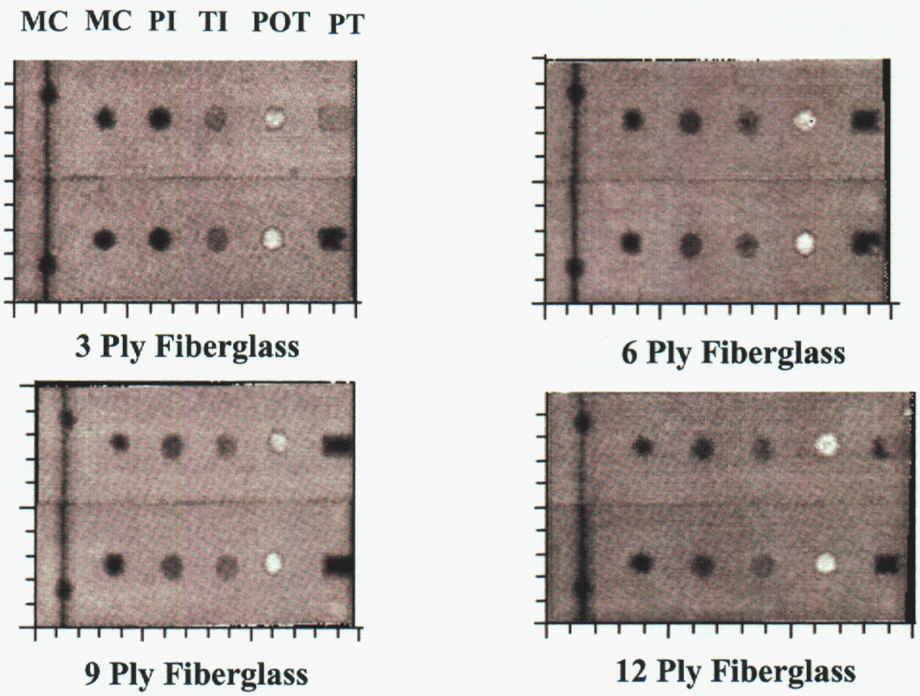


Figure 19: Boeing TTU Scans of 3, 6, 9, and 12 Ply Fiberglass Skins Over 1" Honeycomb

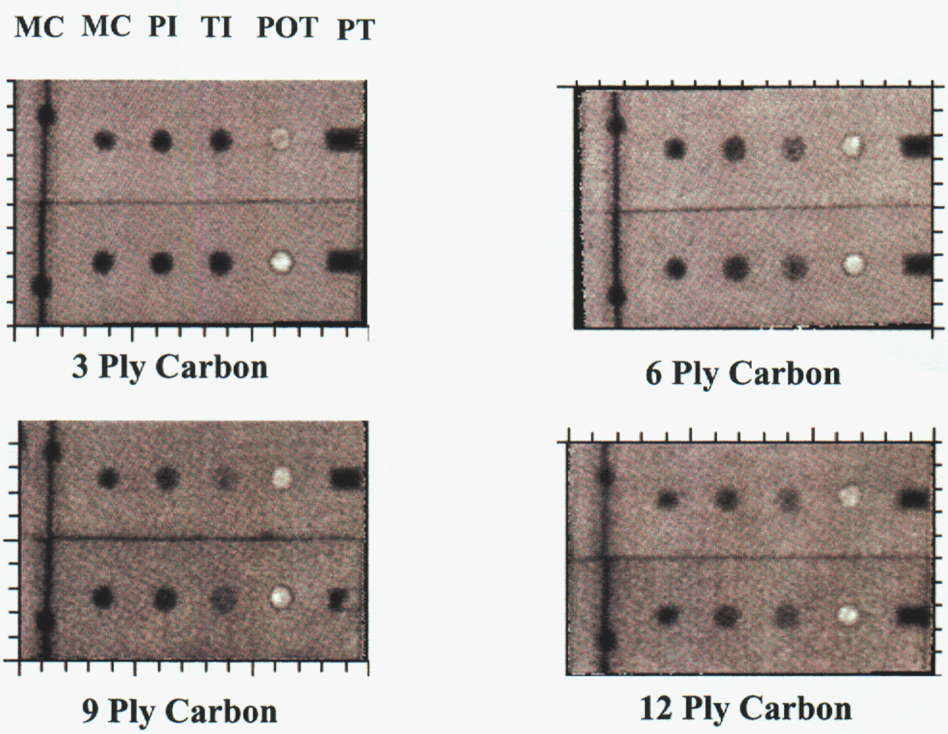


Figure 20: Boeing TTU Scans of 3, 6, 9, and 12 Ply Carbon Skins Over 1" Honeycomb

Subsets of the hand-held S-9 inspections were repeated using an S-9 scanner device. The automated C-scan system eliminated the variability of human probe deployment and provided full-field data. The specimen matrix was: 1) 3 and 12 ply Nomex core with fiberglass or carbon skins, 2) 3 and 12 ply fiberglass core with fiberglass or carbon skins, and 3) Boeing parametric repair panels with scarf repair, potted core areas, disbonds, and delaminations. Figures 21 and 22 show the Sondicator C-scan images where the appropriate areas on the calibration standards are used to produce the equipment set-up for flaw detection in the suite of 64 honeycomb aircraft panels. It can be seen that all flaws were detected in the aircraft panels (ref. Fig. 3 for flaw layout). Other Boeing in-house specimens, including honeycomb structure removed from aircraft, were brought into the validation testing. In all tests, signal-to-noise (S/N) results from the panels indicated acceptable flaw detection in the areas of known flaws.

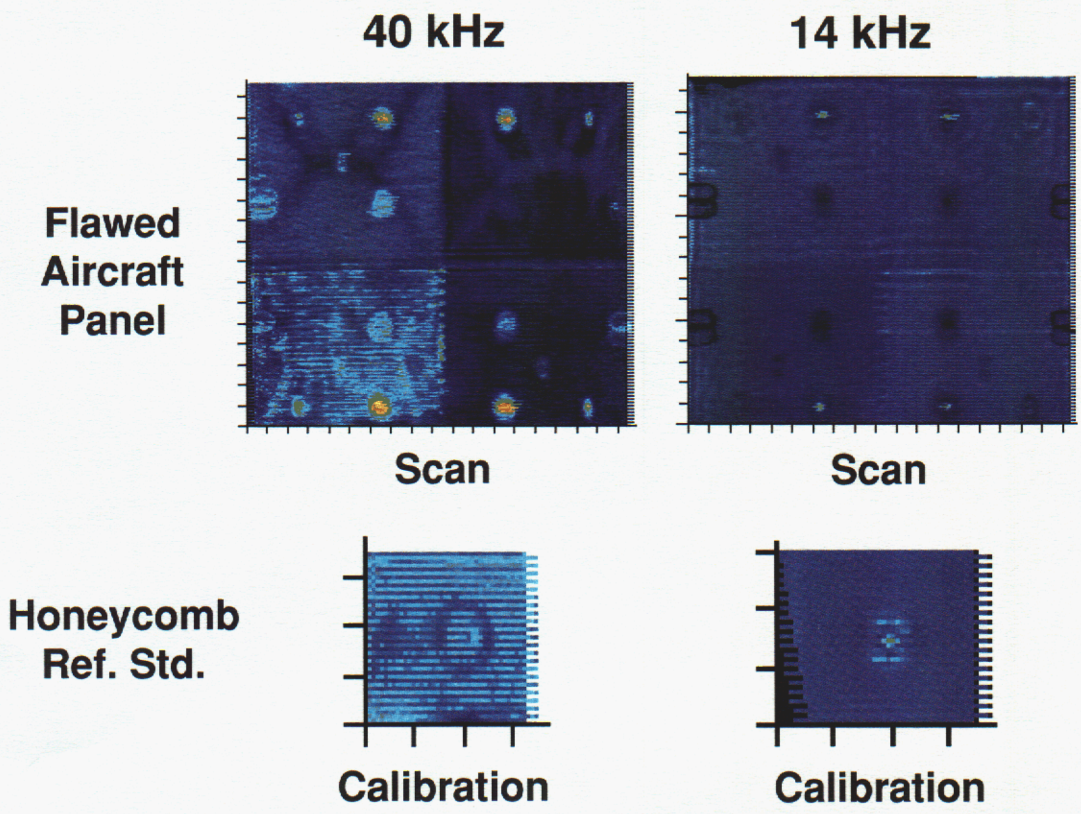


Figure 21: Honeycomb Standard Validation Results from Sondicator C-Scan Inspections (Set of four construction scenarios from Panel #10; 3 ply carbon skin)

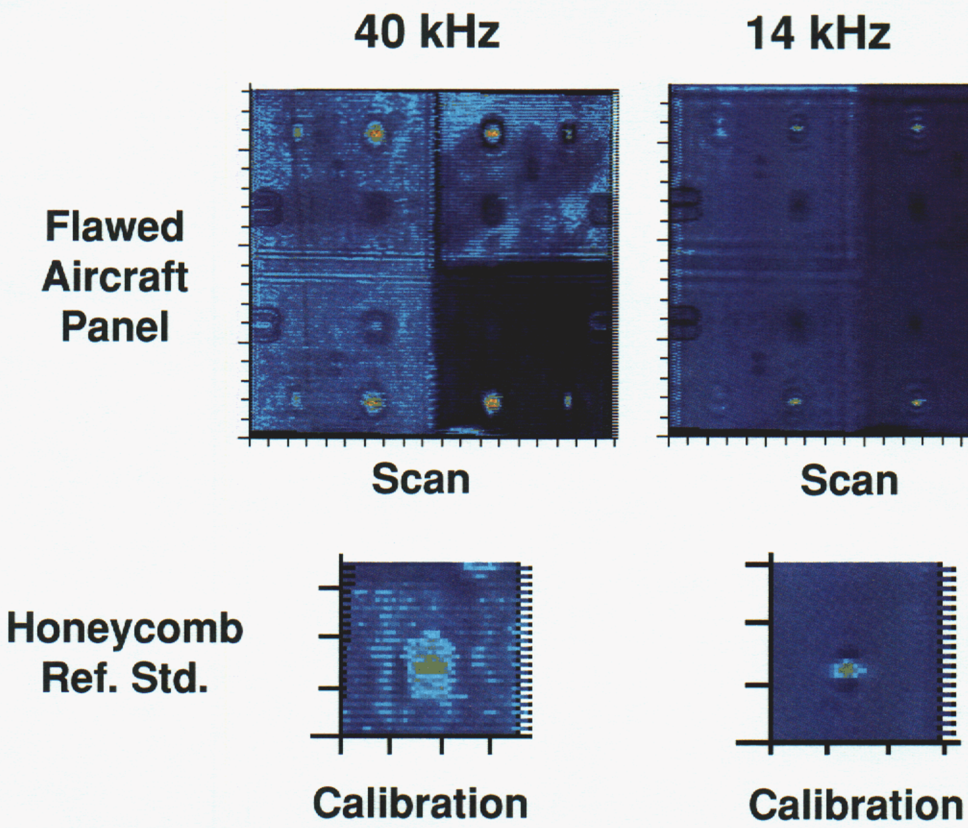


Figure 22: Honeycomb Standard Validation Results from Sondicator C-Scan Inspections (Set of four construction scenarios from Panel #12; 12 ply fiberglass skin)

6.2 BRITISH AEROSPACE AND AIRBUS INDUSTRIES EVALUATIONS

British Aerospace and Airbus Industries also received the specimens and conducted their in-house validation testing using an array of currently-used NDT techniques and equipment. Airbus affiliates conducted the same evaluation exercise as was recently completed by our U.S. team. Inspectors used the prototype reference standards to set up their equipment. They then inspected the set of aircraft test specimens to study flaw detection through the use of the honeycomb standards. If all flaws could be detected (based on 2:1 signal-to-noise ratio), the test results will further support our conclusion that we have adequately captured the important elements in our standards for inspection of composite aircraft structures. Another aspect of the testing involved basic characterization of the honeycomb standards. This exercise assessed the viability of the engineered flaws and looked at overall standard usability. The Airbus affiliates used the same standard operating procedures and flaw detection criteria as those used to inspect an aircraft. The Airbus participants were: British Aerospace, Airbus-United Kingdom, Airbus-Spain, Airbus-Germany, and Airbus-France. This round-robin testing program was carried out as a precursor to Airbus' adoption of the composite reference standards developed in this program.

These tests produced an extensive set of validation testing data from a series of hand-held NDT devices. The equipment applied included the Staveley Sonic 136, Staveley Bondmaster (MIA mode), Zetec Sondicator S9, and Mitsui Woodpecker. Flaw detection results, based on signal-to-noise thresholds, indicate acceptable flaw detection over the range of honeycomb types found on aircraft. Characterization of the standards revealed a repeatable fabrication process and acceptable attenuation levels at the flaw sites.

Airbus tests also highlighted different utilization of the calibration standards based on the technique being deployed. The Bondmaster in MIA mode uses both the flawed and unflawed regions to set up the equipment so an NDI reference (calibration) standard is needed. Conversely, the Woodpecker used the standards simply as a baseline or learning tool. Airbus referred to this as a confidence test. Actual flaw detection was established by using three unflawed points on the panel being inspected to set up the equipment. The S9 device can be applied in either fashion. AANC tests used the standards to establish gains that remained fixed during the inspections. Airbus used the standards as a reference point but adjusted gains on the aircraft panels to optimize output. As a result of this observation, it was decided that there should be a general reference to the standards in the nondestructive testing manuals and that their use should be customized to the NDI technique and device being deployed and the structure being inspected.

Airbus applied the Bondmaster in resonance, MIA, and pitch-catch (LFBT) mode. The resonance inspections were conducted in the same manner as the AANC: NDI reference standards were used to establish a proper gain and then nothing, except rotation, was adjusted when inspecting the aircraft panels. Flaw detection results were similar to those found by our team. Airbus is addressing final data repeatability and manufacturing quality assurance issues in preparation for adoption of the standards. Ultimately, implementation of the standards has come via guidance in the aircraft manufacturers NDT Manuals.

6.3 RELIABILITY CHECK AND FABRICATION PROCESS

The final set of honeycomb standards was produced and beta-site activities were completed. Northwest Airlines' composite shop produced two complete sets of standards and they were characterized by both C-scan and hand-held techniques. In addition a subset of the complete set was manufactured by two different private industry shops. These trial standards were also characterized. Characterization of the standards revealed a repeatable fabrication process and acceptable attenuation levels at the flaw sites. All results indicate that the fabrication specification is able to produce repeatable honeycomb panels.

7. ADOPTION OF HONEYCOMB STANDARDS BY AVIATION INDUSTRY

7.1 SAE AEROSPACE RECOMMENDED PRACTICE AND MODIFICATIONS TO OEM MANUALS

All aspects of the honeycomb reference standard design and production have been formally documented in the Society of Automotive and Aerospace Engineering (SAE) Aerospace Recommended Practice (ARP) 5606 (see Appendix A and Ref. [6]). The ARP includes design drawings, fabrication specifications, certification requirements and quality assurance measures for the standards. This provides the central reference point to control changes and to assure that aircraft manufacturers world wide have access to the latest information. Aircraft manufacturers and airlines will be notified of any modifications to the standards through a revised edition of this ARP. OEM manuals have been modified to include drawings of the standards and a reference to the ARP. Manual revisions also include a reference to recommended fabrication shops to ensure that maintenance depots are provided with consistent and valid NDI standards.

7.2 USE OF HONEYCOMB NDI REFERENCE STANDARDS

The inclusion of the standards in Nondestructive Testing Manuals is being accompanied by guidance on what techniques work well with the standards. It is important to note where a lack of flaw detection in the standards is associated with a limitation of the technique as opposed to a limitation of the standards.

7.3 FABRICATION AND CERTIFICATION OF STANDARDS

All NDI Reference Standards will be certified for use via a through-transmission ultrasonic C-scan inspection. Each honeycomb part will be tested and specific attenuation levels must be achieved at the areas of interest relative to the unflawed areas. Figure 23 shows the use of closed-cell foam tape as a quality assurance device for measuring attenuation levels at the engineered flaw sites and Table 3 lists the required attenuation levels needed for acceptance of the standard. Reference standard manufacturers will be asked to perform this inspection and provide a certificate of conformity on each part. To provide a consistent basis of comparison across multiple production runs and multiple manufacturing shops, a closed-cell foam tape will be included in the TTU certification process. The tape will be placed on the honeycomb specimen and attenuation levels in the tape region will be compared to levels in the unflawed and flawed areas to provide a repeatable Quality Assurance measure.

7.4 CERTIFICATION OF MANUFACTURERS

Two private composite companies - Applied Aerospace Structures Corp. and NDT Engineering - were asked to produce trial honeycomb standards in order to certify their shops. Each company fabricated one 6 ply fiberglass skin and one 6 ply carbon skin standard, each containing both fiberglass and Nomex core. The specimens were then inspected by the AANC and Boeing to assess the manufacturer's capability. All panels compared well with the standards produced to date and all critical areas achieved proper attenuation levels as per the ARP acceptance criteria listed in Table 3.

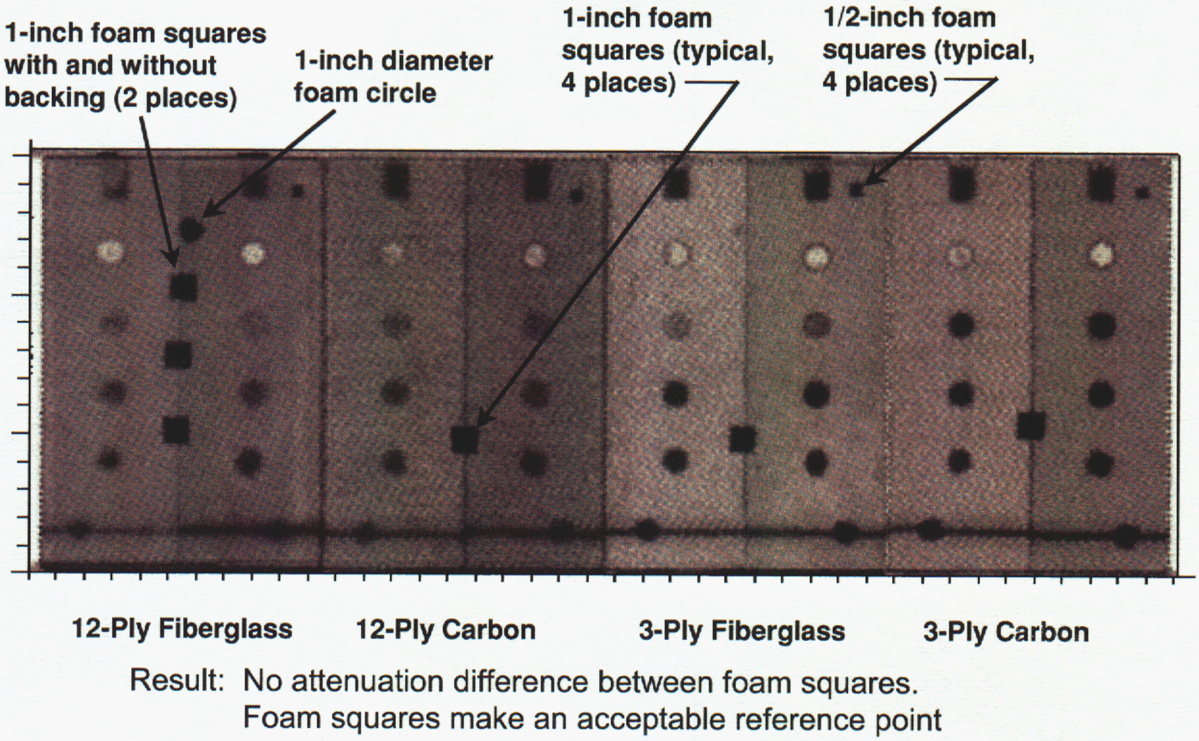


Figure 23: Through-Transmission UT Results Showing Use of Closed Cell Foam as Adequate QA Device in Reference Standard Production

Table 3: Acceptance Criteria for Ultrasonic Inspection of Standards

Reference Standard Location	Acceptance Limits *
Pillow Insert (Interply Delamination)	The ultrasonic attenuation of the Pillow Insert areas must be at least 12dB greater than the attenuation of the Ref. Std. areas without defects.
Machined Core (Disbond)	The ultrasonic attenuation of the Machined Core areas must be at least 12dB greater than the attenuation of the Ref. Std. areas without defects.
Potted Core	The ultrasonic attenuation of the Potted Core areas must be at least 6dB less than the attenuation of the Ref. Std. areas without defects.
Unflawed Area	The ultrasonic attenuation of unflawed areas must be at least 18dB less than the attenuation of the foam tape on the Ref. Std.

**Use a 1 MHz Through-Transmission Ultrasonic (TTU) inspection system.*

PART II: COMPOSITE LAMINATE NDI REFERENCE STANDARDS

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8. INTRODUCTION

8.1 PURPOSE

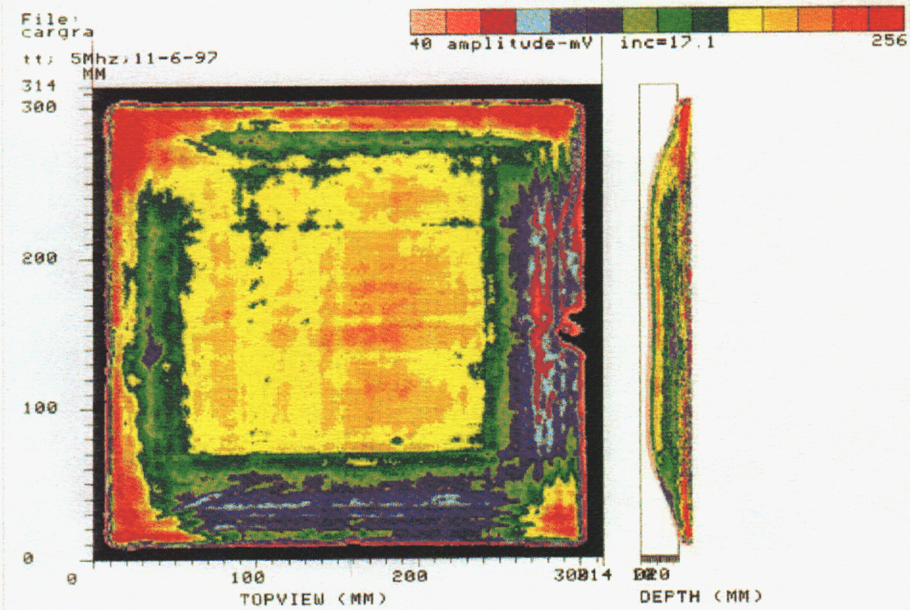
The goal of this effort was to establish a single, generic set of composite laminate NDI reference standards that will accommodate inspections on the full array of fiberglass and carbon laminates found on aircraft. This study took the optimal approach of substituting a single material for both carbon and fiberglass solid laminate inspections. The material had to provide NDI responses similar to both carbon and fiberglass. In addition, in order to improve on existing solid laminate standards, the material had to be inexpensive, reliably manufactured and easy to machine into a solid laminate standard (i.e. plate with multiple thicknesses).

The advantages of industry-wide acceptance of these composite standards include: 1) provides a consistent approach to composite inspections thus improving inspection reliability, 2) reduces standard procurement costs, and 3) aids in future assessments of composite inspection technologies. Specific use of the laminate standards described in the resulting ARP can be achieved through the OEM inspection procedures found in Nondestructive Testing Manuals and Nondestructive Testing Standard Practices Manuals.

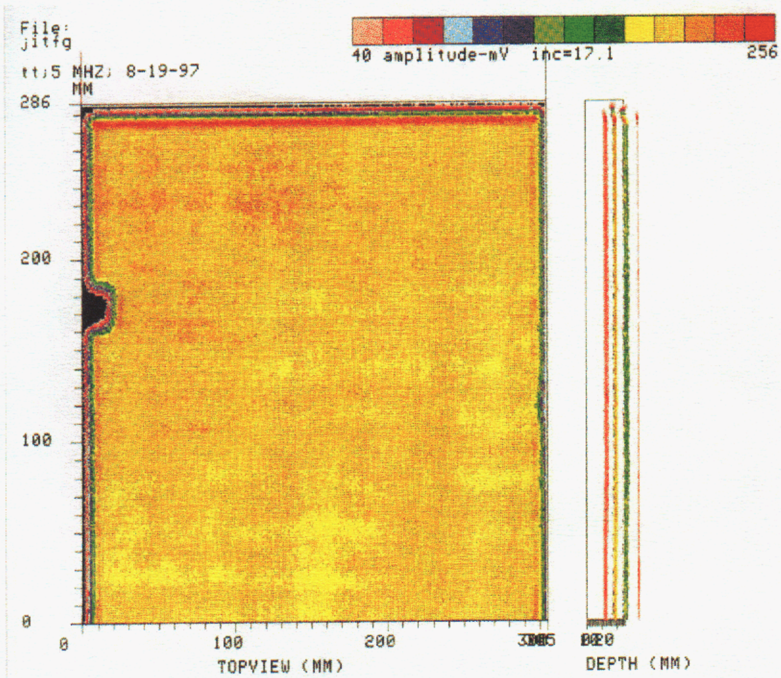
8.2 COMPOSITE LAMINATE STANDARDS BACKGROUND

The current practice of making solid laminate standards out of the same material as the structure being inspected (carbon or fiberglass) results in a lot of scrappage. Figure 24 highlights the difficulty of making thick laminate plates with uniform properties that produce consistent NDI responses. Figure 24 displays two through-transmission ultrasonic C-Scan images that compare the material variations (attenuation) in a thick carbon laminate and a thick plate of G11 material. The variation in attenuation of some of the laminates, such as the 24 dB signal attenuation shown in Fig. 24A, is so severe that the part must be rejected. This fact, coupled with the cost of the laminate material and the labor needed to lay up all of the plies, makes these type of standards very expensive. The alternative presented here uses a G11 Phenolic material to adequately mimic the NDI response observed in carbon and fiberglass laminates. G11 Phenolic is manufactured at high temperatures and pressures to produce a very uniform material as shown in Fig. 24B. Various thicknesses, simulating laminates with different numbers of plies, can be produced in the G11 material by a simple machining process.

The results from the solid laminate reference standard effort are formally documented in the SAE Aerospace Recommended Practice (ARP) 5605 (see Appendix B and Ref. [7]). The purpose of ARP 5605 is to describe the design and production of solid composite laminate calibration standards to be used in ultrasonic, resonant, and tap test NDI equipment calibration for accomplishment of damage assessment and post-repair inspections. These standards have been adopted by aircraft manufacturers within procedures contained in their Nondestructive Testing Manuals. Depending on the nature of the inspection, it may be necessary to compensate for variations in material properties through the use of correction factors or by adjusting for these differences on the part or structure being inspected. When using these standards, consideration must be given to surface coatings such as paint or lightning protection plies.



A. Signal Variations Observed Across a 0.5" Thick Carbon Laminate Plate



B. Signal Variations Observed Across a 1.0" Thick G11 Phenolic Plate

Figure 24: Through-Transmission UT C-Scan of a Thick Carbon Laminate (top) Shows Significant Material Variations (Attenuation); Comparable C-Scan of 1" Thick G11 Plate (bottom) Shows a Very Uniform Material with Less Than 2 dB Variation in Attenuation

8.3 COMPOSITE LAMINATE INSPECTION METHODS

In addition to the inspection methods described in Section 1.3, pulse-echo ultrasonics is useful for inspecting solid laminate composite structures. In Pulse-Echo Ultrasonic (P-E UT) inspections, short bursts of high frequency sound waves are introduced into materials for the detection of surface and subsurface flaws in the material. The sound waves travel through the material with some attendant loss of energy (attenuation) and are reflected at interfaces. The reflected beam is displayed and then analyzed to define the presence and location of flaws.

Ultrasonic testing involves one or more of the following measurements: time of wave transit (or delay), path length, frequency, phase angle, amplitude, impedance, and angle of wave deflection (reflection and refraction). In most pulse-echo systems, a single transducer acts alternately as the sending and receiving transducer. If the pulses encounter a reflecting surface, some or all of the energy is reflected and monitored by the transducer. The reflected beam, or echo, can be created by any normal (e.g. in multi-layered structures) or abnormal (flaw) interface. Figure 25 is a schematic of the pulse-echo technique. It shows the interaction of UT waves with various interfaces within a structure and the corresponding A-scan waveforms that are displayed on an ultrasonic inspection instrument. Sometimes it is advantageous to use separate sending and receiving transducers for pulse-echo inspection. The term pitch-catch is often used in connection with separate sending and receiving transducers. The degree of reflection depends largely on the physical state of the materials forming the interface. Cracks, delaminations, shrinkage cavities, pores, disbonds, and other discontinuities that produce reflective interfaces can be detected. Complete reflection, partial reflection, scattering, or other detectable effect on the ultrasonic waves can be used as the basis of flaw detection. In addition to wave reflection, other variations in the wave that can be monitored include: time of transit through the test piece, attenuation, and features of the spectral response.

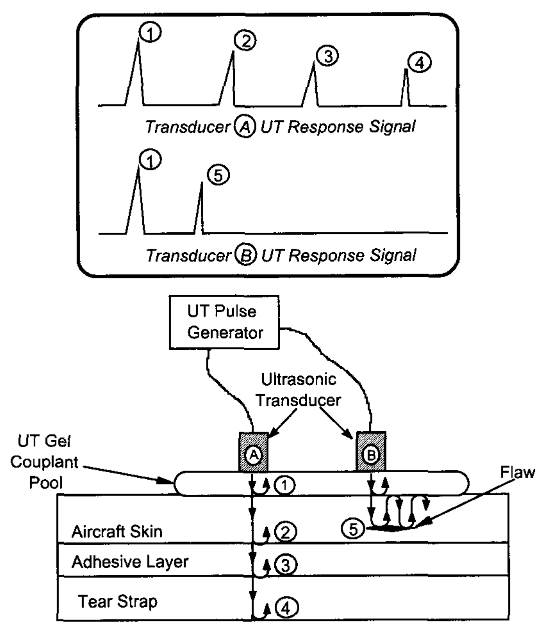


Figure 25: Schematic of Pulse-Echo Ultrasonic Inspection and A-Scan Signal Showing Reflection of UT Waves at Assorted Interfaces

Scanning Ultrasonics - It is sometimes difficult to clearly identify flaws using ultrasonic A-Scan signals alone. Small porosity pockets commonly found in composites, coupled with signal fluctuations caused by material nonuniformities can create signal interpretation difficulties. Significant improvements in disbond and delamination detection can be achieved by taking the A-Scan signals and transforming them into a single C-Scan image of the part being inspected. C-scans are two-dimensional images (area maps) produced by digitizing the point-by-point signal variations of an interrogating sensor while it is scanned over a surface. A computer converts the point-by-point data into a color representation and displays it at the appropriate point in an image. Specific "gates" can be set within the data acquisition software to focus on response signals from particular regions within the structure. C-Scan area views provide the inspector with easier-to-use and more reliable data with which to recognize flaw patterns.

9. USE OF KEY MATERIAL PROPERTIES TO SELECT A GENERIC LAMINATE FOR NDI

9.1 CHARACTERIZATION OF EXISTING LAMINATE STANDARDS

Through-transmission and pulse-echo ultrasonics, along with resonance testing, were applied to the series of existing Boeing, Douglas, and Airbus laminate standards (step wedges of various materials at different thicknesses) in order to measure the key velocity, acoustic impedance, and attenuation characteristics in the laminates. The results are tabulated in Table 4.

Table 4: Important Material Properties for Candidate Laminate Standard Materials

Material	Velocity in/μs (mm/μs)	Density g/cm^3	Acoustic Impedance g/cm^2·μs	Relative Attenuation*
Carbon Graphite Boeing Std ST8870 (BMS 8-212)	0.1218 (3.070)	1.589	0.488	-
Carbon Graphite Boeing Std ST8871 (BMS 8-276)	0.1150 (2.912)	1.589	0.463	-
Fiberglass (50 V%)	0.1150 (2.912)	1.917	0.605	20 dB (0.2" th.) 30 dB (0.5" th.)
Boron-Epoxy (50 V%)	0.1310 (3.317)	1.920	0.639	Not Measured
Ivory	0.1185 (3.000)	2.170	0.653	Not Measured
Hysol Potting Material EE4183	0.1010 (2.562)	1.518	0.390	10 dB (0.2" th.)
Phenolic [United Airlines supply]	0.1100 (2.873)	No Data	No Data	12 dB (0.2" th.) 18 dB (0.5" th.)
Phenolic G7	0.0834 (2.110)	1.700	0.358	Not Measured
Phenolic G9	0.1474 (3.730)	1.950	0.727	Not Measured
Phenolic G10	0.1193 (3.020)	1.850	0.559	4 dB (0.2" th.) 9 dB (0.5" th.)
Phenolic G11	0.1158 (2.930)	1.850	0.541	4 dB (0.2" th.) 9 dB (0.5" th.)
Phenolic LE	0.1047 (2.650)	1.320	0.350	Not Measured
Phenolic XXX	0.1071 (2.710)	1.300	0.352	Not Measured
Zero Impedance Material	0.0843 (2.141)	1.240	0.2655	Not Measured
Generic Material Targets	**0.1150 (2.911)**	**1.6 - 1.8**	**0.48 - 0.54**	< 10 dB

* *As compared with Boeing carbon composite step wedge ST8870, 0.2" th. and Boeing carbon composite step wedge ST8871, 0.5" th.*

a. Pulse-Echo Testing (ultrasonic velocity measurements) - Longitudinal velocity data was acquired using 1 MHz, 2.25 MHz, and 5 MHz transducers. The velocity data was very consistent across each step wedge and even similar from one material to another. The maximum difference between the minimum and maximum velocities for all existing OEM standards (see first three items in Table 4) including fiberglass and carbon materials was less than 10%. The velocities ranged from 0.115 in/µs to 0.122 in/µs. These results are logged in Table 4 and produced the target values shown for our generic material. Based on the velocity results, it was determined that for velocity-based equipment, it may be sufficient to use a single laminate standard. The standard should be made from a material with a median velocity of 0.115 in./µs (see target levels at bottom of Table 4).

b. Resonance Testing - Velocity measurements alone do not allow for proper resonance equipment set-up. Furthermore, resonance testing requires that the equipment be set-up on laminates with similar thickness to the part being inspected. Thus, the necessary laminate reference standards should have the appropriate material property. The key property is acoustic impedance, Z, where ρ = density and

$$Z = \rho \text{ X Velocity} \tag{2}$$

In this case, the standard should be made from a material with an acoustic impedance around 0.5 $g/cm^2 \cdot \mu s$ (see target levels at the bottom of Table 4).

c. Attenuation Data - A significant number of the attenuation values varied substantially in a single step wedge (current OEM reference standards). Numerous factors affect attenuation measurements and this parameter is difficult to use to correlate one laminate with another. In fact, some carriers indicated that they use laminate standards to set up their equipment (functionality) but not to establish flaw call "levels." Attenuation in the laminate standards doesn't exactly represent the actual part on the aircraft. Inspectors base flaw calls on consistency across the part being inspected (in-situ measurements determine appropriate signal levels). However, this parameter does provide a basis of comparison with existing laminate standards. The goal was to match the attenuation of the existing laminates and not induce additional attenuation through the introduction of a new generic material. For this parameter it was decided to search for material that would lie halfway between the fiberglass attenuation (20-30 dB relative to carbon) and an exact match with carbon. In order to accommodate inspections through thick laminates (0.25" - 0.5" thick), the target goal shown in Table 4 is for an attenuation level of 10 dB or less relative to the existing carbon step wedges.

9.2 IDENTIFICATION OF A CANDIDATE MATERIAL: G11 PHENOLIC

Based on the above observations, a search was performed to locate a material with the appropriate properties. Other desirable attributes were that the material be inexpensive, easy to machine, and able to be reliably produced. Table 4 lists candidate materials along with the data from the current carbon and fiberglass NDI Reference Standards. The material search, involving ultrasonic velocity, acoustic impedance, and relative attenuation measurements, identified G11

Phenolic as the best candidate for a generic solid laminate reference standard material. Testing determined matches in velocity and acoustic impedance properties, as well as, low attenuation relative to carbon laminates. Through-transmission ultrasonic inspections of G11 showed that it could be manufactured as very pure material with very little porosity. As shown in Figure 24, there was basically no porosity measured. Ultrasonic C-scans showed less than 2 dB in signal variation across the entire 12" X 12" area.

9.3 DESIGN FEATURES OF G11 SOLID LAMINATE STANDARD SET

A series of solid laminate standards were produced from the G11 material to conduct validation testing. Figure 26 shows the design drawings while Figure 27 contains a photo of four of the laminates in the five-standard set. The basic design approach is to machine flat-bottomed holes in a plate that is large enough to accommodate scanner heads. This plate design will be less susceptible to breakage than the existing wedge specimens. The key issues addressed by the designs are as follows: 1) protection against moisture ingress - extensive exposure to water submersion showed that water absorption is not a problem with G11 material, 2) locating the probe - the location of each skin thickness is identified on the laminates to allow for proper positioning of the transducer and each thickness is labeled, 3) surface finish - the surface finish was improved via a lapping process to produce more consistent responses from the transducers, 4) size and ease of handling - the set of 30 thicknesses is distributed over five different plates, and 5) one ply resolution - the thinnest skin was placed at 0.007" thick to closer represent the thickness of one ply.

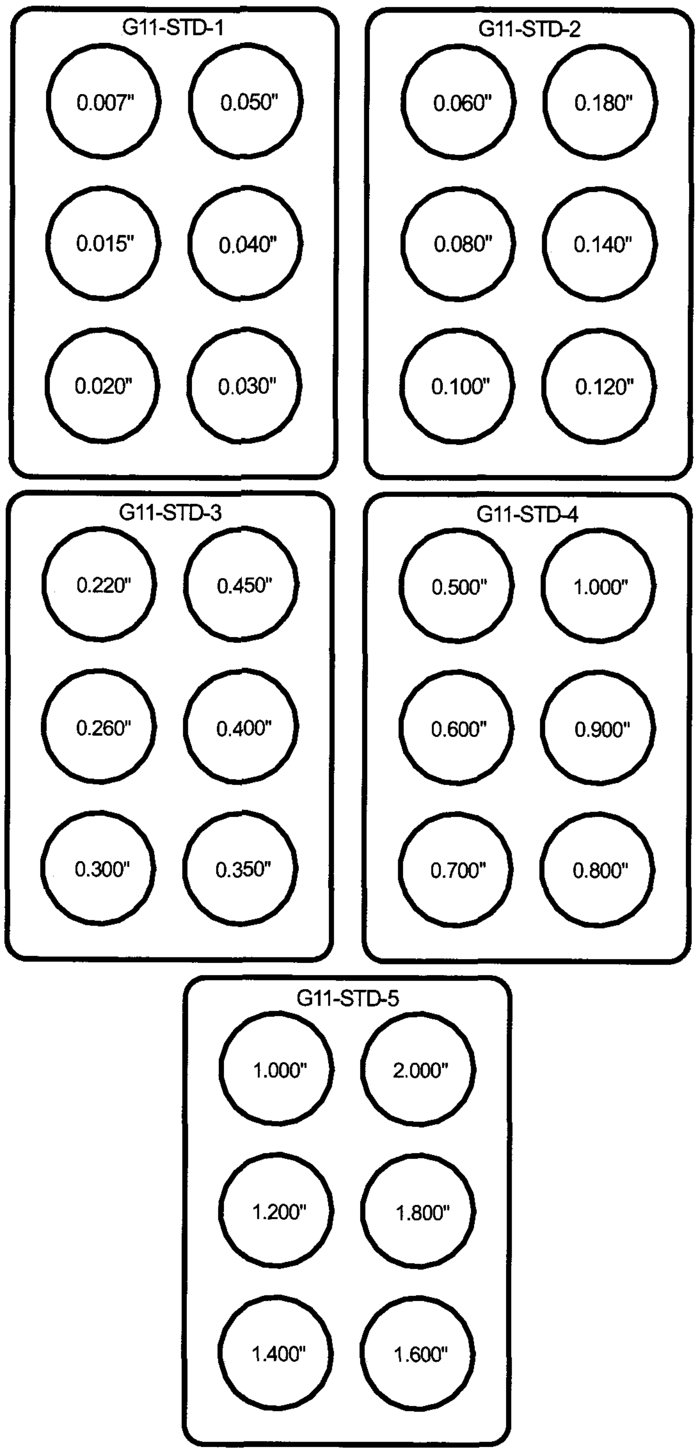


Figure 26: G11 Solid Laminate NDI Reference Standard Set (numbers in circles represent skin thickness, in 1/1000", at flat bottom holes; each plate is 4" W X 5.75" H)

Figure 27: Front and Back Photos of G11 Solid Laminate NDI Reference Standard Set

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10. VALIDATION TESTING USING HAND-HELD ULTRASONIC TECHNIQUES

10.1 VALIDATION TEST PLAN TO COMPARE G11 PHENOLIC STANDARDS WITH EXISTING OEM STANDARDS

Following is a test plan that was produced to acquire quantitative NDI signal data for comparing G11 phenolic standards with existing OEM carbon graphite and fiberglass laminate standards. A series of validation tests were performed by the OEM, airline, and AANC team members. Baseline testing was performed on different sets of existing laminate standards in order to compare responses (i.e. assess uniformity and response variations) from duplicate standard sets. Baseline testing was also performed on similar laminate structures found on different aircraft in order to determine the envelope of response variations that can be expected in the field.

These tests produced the following data: 1) comparisons between the G11 laminate standards and existing laminate standards, 2) comparisons between inspections performed on aircraft structures using the G11 and the carbon or fiberglass standards, 3) comparisons between similar existing laminate standards to assess their variations, and 4) comparisons between similar laminate aircraft structure to assess their variations. The first two tests established how well the G11 standards match the results produced by existing carbon and fiberglass standards. The latter two tests established how close this match must be to accommodate field inspections (i.e. the allowable deviation between the existing standards and proposed G11 standards given the variations that occur naturally in similar aircraft laminates). The definitions of the validation tests follow while the results from these tests are described in Section 10.2.

1. Comparisons of Inspection Results Via Hand Held Ultrasonic Techniques

    a. Resonance Inspection (High Frequency Bond Test Device) - Resonance response curves, or thickness loci spiral curves, can be produced to compare the G11 phenolic with existing carbon and fiberglass standards. The inspection process is as follows: 1) null the equipment on the thickest region of the G11 standard that you would like to include in the spiral curve (current data sets use 0.260" th.), 2) set the gain and rotation so that the data down to the minimum thickness will lie within the screen (current data sets use 0.007" th.), 3) establish the thickness loci spiral curve for the G11 material using each of the thickness steps without changing any equipment settings and, 4) produce a thickness loci spiral curve from an existing reference standard of interest. In some cases, the existing standards may not have enough thickness variations to fully populate a smooth spiral curve. To accommodate the comparisons, discrete points can be plotted to show how they compare with the G11 curve.

    b. Pulse-Echo - Pulse-echo ultrasonic inspections can be used to make comparisons between the G11 material and existing laminates in two separate ways. First, the instrument can be used to determine the thickness of a laminate test article. Comparisons can be made between thickness values produced by a calibration on the G11 material and thickness values produced by a calibration on a fiberglass or carbon laminate standard. In this type of testing note that the velocity parameters will be slightly different for G11, carbon, and fiberglass as per Table 4.

A second use of pulse-echo equipment would be to study the A-scan waveforms and compare the attenuation and damping in the pulses reflected from the back surface. Airbus has suggested this test since inspectors may look at changes in this back surface pulse to infer the presence of a flaw. In this type of testing note that measurements have shown a 4 - 9 dB signal attenuation in the G11 material relative to carbon as per Table 4. However, signals from the G11 material should be stronger than in fiberglass since fiberglass is more attenuative.

c. Other Conventional NDI Techniques Applicable to Solid Laminate Inspections - Although other NDI techniques are used to a lesser extent on solid laminates, additional data may be obtained from Mechanical Impedance Analysis (MIA) or Low Frequency Bond Test (LFBT) devices. Again, validation testing can take two different forms: 1) comparisons between equipment responses produced in existing composite laminate standards and equipment responses in the G11 laminate standards, and 2) comparisons between aircraft structure inspections supported by calibration on the G11 standards and aircraft structure inspections supported by calibration on carbon and fiberglass standards.

2. Scanning Ultrasonics - In order to eliminate data variations stemming from hand-held probe deployment, automated C-scan inspections can be performed. Either resonance or pulse-echo ultrasonics can be used. The sensor can be nulled over the laminate thickness of interest on the G11 material. Without renulling the instrument, a scan can then be made on all other laminate standard panels that contain a matching thickness. This will produce a C-scan image and comparative raw data counts for each of the different materials.

3. Field Testing - This section of the test plan uses the same techniques as those described above, however, a separate discussion is provided to emphasize the importance of assessing the field performance of G11 relative to existing standards. The tests outlined below produce data comparing inspections performed on aircraft structures using the G11 NDI Reference Standards and inspections performed on aircraft structures using existing carbon or fiberglass standards. Furthermore, data will be acquired to establish the naturally occurring variation in similar aircraft laminate structures. This latter data will determine what type of resolution can realistically be demanded from the G11 standards.

a. Thickness Mapping on Aircraft Structure - This testing can be performed with resonance or pulse-echo techniques as described above. The key in this type of testing is to have aircraft structure with well-known laminate thicknesses or flaw depths. A G11 thickness loci spiral curve (resonance set-up) and pulse-echo readings can be compared with depth and thickness predictions from existing laminates. Both of these can then be compared to known values for the aircraft structure.

b. Laminate Variations on Common Aircraft Structures - Another test approach is to compare responses from existing standards with those from multiple, similar aircraft structure. These type of tests will measure the envelope of response variations that can be expected in the field. In this test series, resonance inspections are performed by setting up the equipment on existing laminate standards. This will produce a spiral curve from which laminate thickness mapping can be performed. Next a series of laminate aircraft structures (elevators, rudders, etc.) are inspected and the locations of the dots on the resonance screen are plotted relative to the calibration curve. This process is repeated on duplicate structures found on different aircraft producing a spread of data points for each common thickness region. Once again, the key in this test approach is to have well-documented aircraft structure with known laminate thicknesses.

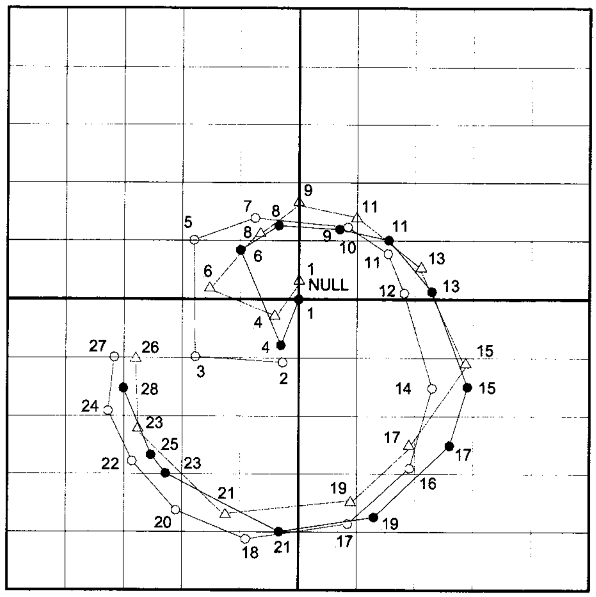
c. Use by Aircraft Inspectors - In addition to the inspections performed by team members, it is important to allow maintenance depot inspectors to try out the G11 standards. These type of tests will be very similar to the ones described in items (b) above. Except, in this case the G11 standards will be used for calibration in order to assess aircraft laminate thickness mapping using this calibration curve. Inspectors can also conduct pulse-echo comparisons as described in item (a) above.

10.2 VALIDATION TESTING USING RESONANCE INSPECTION

In this validation testing, resonance response curves were produced. The inspection process was as follows: 1) the equipment was nulled on the thickest region of the G11 standard that was included in the spiral curve, 2) the gain and rotation was set so that the data down to the minimum thickness could be plotted within the screen, 3) the thickness loci spiral curve for the G11 material were established using each of the thickness steps without changing any equipment settings and, 4) produce a thickness loci spiral curve from an existing reference standard of interest was produced to form a basis of comparison.

Resonance response curves were obtained for high frequency (314 KHz) and low frequency (156 KHz) inspections over a range of high (12 - 14 dB), medium (9-10 dB), and low (6-7 dB) gains. High frequency inspections were used to measure the Bondmaster response over the thickness range of 0.010" to 0.250" while low frequency inspections measured the Bondmaster response over the thickness range of 0.050" to 0.600". For the comparison between carbon, fiberglass, and G11 phenolic, a null point was taken only on the G11 phenolic. Subsequent measurements were taken on the carbon and fiberglass without renulling the instrument. This gives an indication of the response variation between the different materials in specific thickness ranges with setup parameters based on G11. Figures 28-31 show the results from the high frequency inspections and compare the fiberglass and carbon response curves to the G11 material. These figures show that even at high gain, the "spiral" curves are closely clustered. It can be seen that the G11 spiral curves compare even better with fiberglass. This is reasonable since the attenuation and acoustic impedance values are almost identical.

Figure 28 provides an overall view comparing existing fiberglass and carbon standards to G11. The resonance response curves from the G11 phenolic prototype standard were very similar to the resonance response curves measured on the existing carbon and fiberglass laminates. Although the G11 spiral resonance curves do not exactly align with those obtained from the carbon standards, resonance tests on three different sets of carbon composite standards showed that variability across "similar" standards was the same as the variability observed between G11 and carbon or fiberglass (see also Section 10.3). Figure 29 isolates the comparison between G11 and Boeing carbon standards while Figure 30 isolates the comparison between G11 and fiberglass standards. Figure 29 also contains the responses produced by the existing Airbus solid laminate standards. In Fig. 29, responses from the Boeing carbon standards are underlined, responses from the Airbus carbon standards are listed in parenthesis, and responses from the G11 standards are unmarked. Finally, Figure 31 compares the G11 standards with the responses obtained from OEM Embraer carbon standards. It can be seen that the G11 standards provide a very close match with the existing carbon and fiberglass standards and the resonance thickness maps are within approximately one ply of each other.

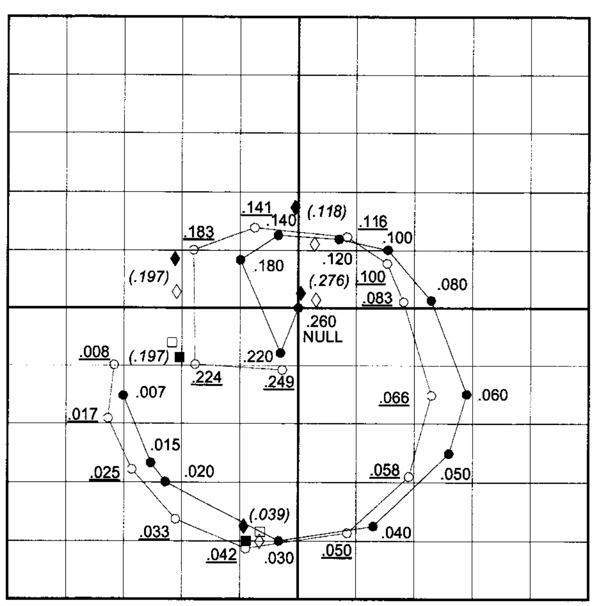


1. Nulled on 0.260" thick G11 phenolic with covercoat.
2. Setup:
   a) Gain=12dB
   b) Frequency = 313KHz
   c) Rotation = 248°
   d) H pos = 50%
      V pos = 50%
3. Thickness range is from .007" thru 0.260".

G11 With Covercoat
Carbon Plain Weave
Fiberglass Step Plate

Hole No.	Thk.	Hole No.	Thk.
1	.260	15	.060
2	.249	16	.058
3	.224	17	.050
4	.220	18	.042
5	.183	19	.040
6	.180	20	.033
7	.141	21	.030
8	.140	22	.025
9	.120	23	.020
10	.116	24	.017
11	.100	25	.015
12	.083	26	.010
13	.080	27	.008
14	.066	28	.007

Figure 28: Comparison of G11 Phenolic with Existing Carbon and Fiberglass Standards

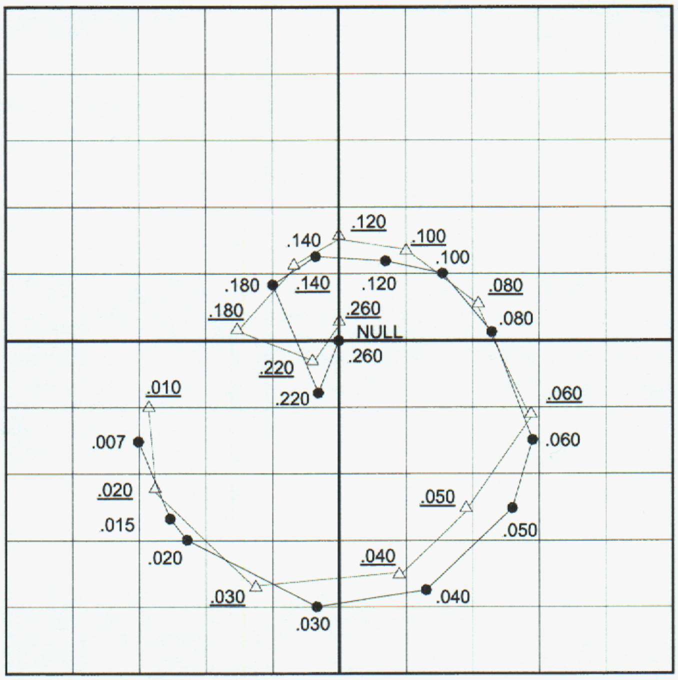


Notes:

1. Nulled on 0.260" thick G11 phenolic with covercoat.
2. Setup:
   a) Gain=12dB
   b) Frequency = 313KHz
   c) Rotation = 248°
   d) H pos = 50%
      V pos = 50%
3. Thickness range is from .007" thru 0.260".

G11 With Covercoat .XXX
Carbon Plain Weave .XXX
Airbus Carbon Fiber Tape (Tape Side)
Airbus Carbon Fiber Tape (Paint Side)
Airbus Carbon Fiber Fabric (Fabric Side)
Airbus Carbon Fiber Fabric (Paint Side)

Figure 29: Direct Comparison of G11 Material with Boeing and Airbus Carbon Laminate Standards



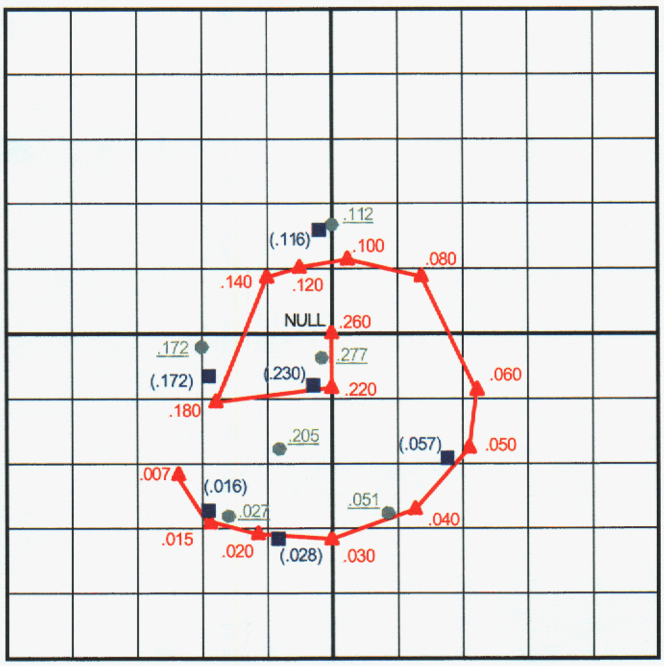
Notes:

1. Nulled on 0.260" thick G11 phenolic with covercoat.
2. Setup:
   a) Gain=12dB
   b) Frequency = 313KHz
   c) Rotation = 248°
   d) H pos = 50%
      V pos = 50%
3. Thickness range is from .007" thru 0.260".

● G11 With Covercoat .XXX

△ Fiberglass Step Plate .XXX

Figure 30: Close-Up View Comparing G11 Material with Fiberglass Standards



Notes:

1. Nulled on 0.260" thick G11 phenolic with covercoat.
2. Inspection Device:
   a) Bondascope 2100 bondtester
3. Setup:
   a) Frequency = 327.1KHz
4. Thickness range is from .007" thru 0.260".

G11 Phenolic Laminate Standard .XXX

Embraer standards LSP (carbon fiber fabric flat laminates) .XXX

Embraer standards LSE (carbon fiber fabric step wedge) (.XXX)

Figure 31: Direct Comparison of G11 Material with Embraer Carbon Laminate Standards

10.3 LAMINATE VARIATIONS OBSERVED IN THE FIELD

As part of the quantification of the G11 performance, it is essential to understand the variations that occur naturally in similar laminates. This will allow us to establish a requirement for the match between G11 and existing laminate standards and to answer the question: "Does the variation between G11 and existing laminate standards fall within the noise level of an inspection or expected deviations between laminate standards and actual aircraft structure?"

These data comparisons are shown in Figures 32 - 34 where specimens made of identical or similar materials are compared against each other to show variations that might occur in the fabrication process (e.g. cure pressure, temperature, etc.). Instrumentation setup for these data included nulling on each individual material. This provides some perspective for the resonance inspection data and allows us to better assess the spread observed in Figures 28 - 31. Figure 32 shows resonance response curves comparing the Boeing uniaxial step wedge with the carbon graphite prototype standard (BMS 8-276) produced by NDT Engineering for this study. Most of the common thickness points plotted close together, however, data spreads similar to the G11-to-carbon comparisons were observed. A Boeing carbon standard and a McDonnell Douglas carbon standard are compared in Figure 33 to show the variations that are present even though the materials (carbon graphite - plain weave) and step thicknesses are the same.

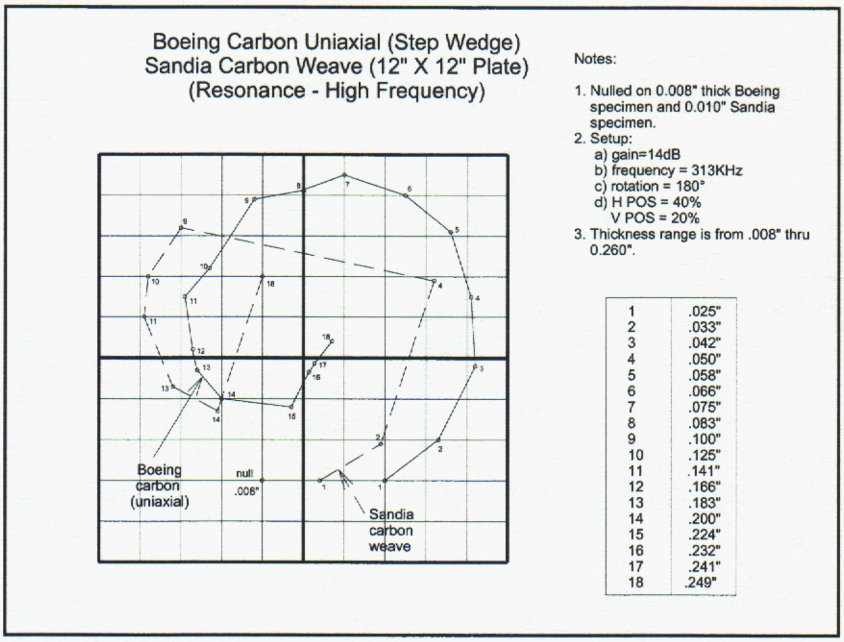
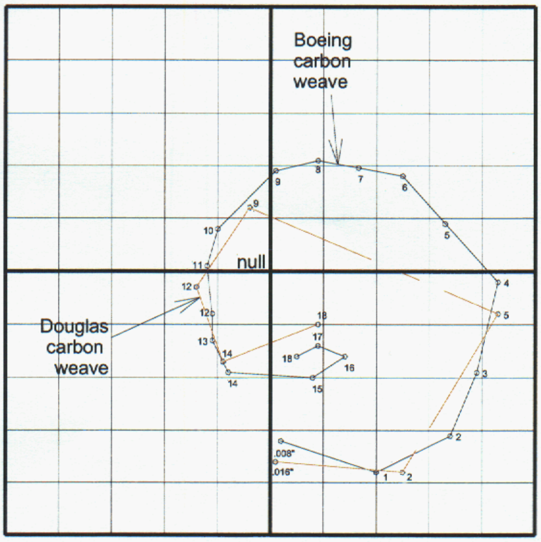


Figure 32: Comparison of Resonance Response Curves for Similar Carbon Reference Standards

Figure 34 compares the response curves from three similar carbon graphite (plain weave) step wedge specimens that were produced by United Airlines' composite shop. The specimens were produced with the intent of simulating the porosity, surface roughness, and irregularities of actual aircraft structure. The irregularities would typically be the result of variations in the fabrication process. These variations, within allowable tolerances, can include parameters such as cure pressure, cure temperature, debulk steps, and other manufacturing specifications. Again, the response is different on each laminate and an envelope of response curves is generated.

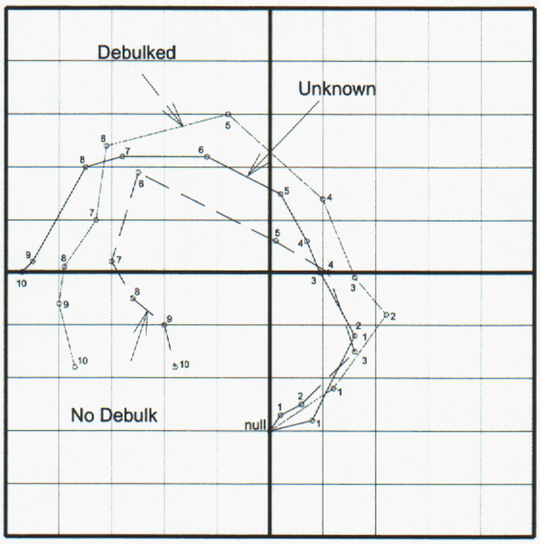


Notes:

1. Nulled on 0.260" thick G11 phenolic.
2. Setup:
   a) gain=12dB
   b) frequency = 313KHz
   c) rotation = 180°
   d) H POS = 50%
      V POS = 50%
3. Thickness range is from .008" thru 0.260".

Figure 33: Comparison of Resonance Response Curves from Boeing and Douglas Carbon Standards

Carbon Weave
Comparison Samples
(Resonance - High Frequency)



Notes:

1. Nulled on 0.020" thick carbon weave section.
2. Setup:
   a) gain=14dB
   b) frequency = 313KHz
   c) rotation = 150°
   d) H POS = 50%
      V POS = 20%
3. Thickness range is as follows:

null	0.020"
1	0.030"
2	0.040"
3	0.050"
4	0.060"
5	0.070"
6	0.080"
7	0.090"
8	0.100"
9	0.110"
10	0.120"

Figure 34: Variation in Resonance Response Curves for Similar Carbon Laminate Aircraft Structure

Additional tests were completed to understand the variations that occur naturally in similar aircraft laminates. This allowed us to establish a firm requirement for the match between G11 and existing laminate standards. The first set of tests determined the range of responses for similar materials of common thickness. From this data it was possible to develop an envelope of spiral curves (thickness loci plots) using similar solid laminate structures on different aircraft. Resonance inspections were performed by setting up the equipment on existing standards (basis of comparison) and then inspecting a series of aircraft elevator and rudder structures. These structures contain a number of different laminate thicknesses and allowed us to produce at least a portion of the thickness loci curve. If the G11 response curve lies within the envelope of curves found in the field, then we will have quantified G11 as an acceptable match to support laminate inspections.

Results from such tests on actual aircraft structure are shown in Figures 35 - 39. The Fig. 35 plot contains resonance response curves obtained from similar composite laminate structure on 767 aircraft. The data was acquired as follows. First, the existing Boeing carbon laminate standard was used to set up the equipment and establish the thickness loci shown by the solid line. Then, without changing the equipment settings, a series of carbon laminate structures on 767 aircraft were inspected. Individual data points of the resonance response at different laminate thicknesses are shown alongside the calibration curve. It can be seen that there are response variations even for common thicknesses on common structures.

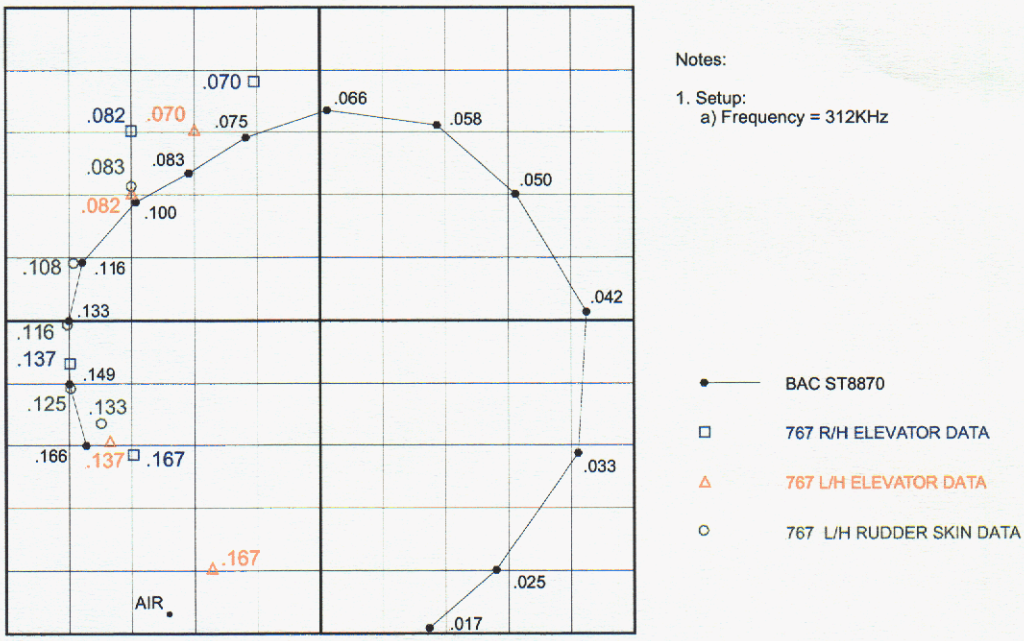


Figure 35: Laminate Response Variations Observed in the Field on Common Aircraft Structure

Additional inspection data similar to Fig. 35 are shown in Figures 36 to 39. These plots contain resonance response curves obtained from similar composite laminate structure on 757 aircraft. Again, existing Boeing carbon laminate standards were used to set up the equipment and establish the thickness loci shown by the solid circles in the figures. Then, without changing the equipment settings, a series of carbon laminate structures on 757 aircraft were inspected. *It can be seen that this data also shows response variations in the laminate structures even for common thicknesses on common structures. Thus, it is not appropriate to require exact matches between the G11 standards and existing laminate standards since Figs. 35 - 39 show that there is some amount of variation between like structures in the field. This data produced a realistic envelope of comparison to demonstrate the ability of G11 to adequately match aircraft laminate responses.*

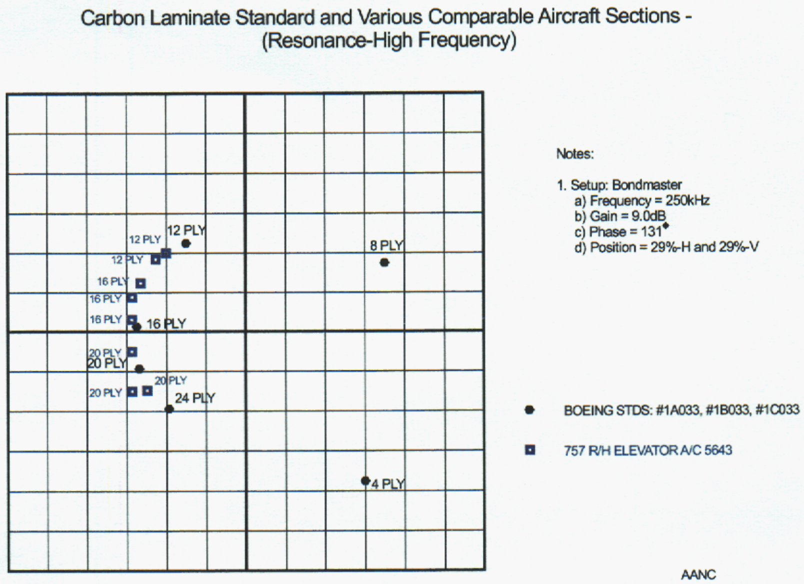


Figure 36: Laminate Response Variations Observed in the Field on a 757 R/H Elevator Structure – Aircraft #1

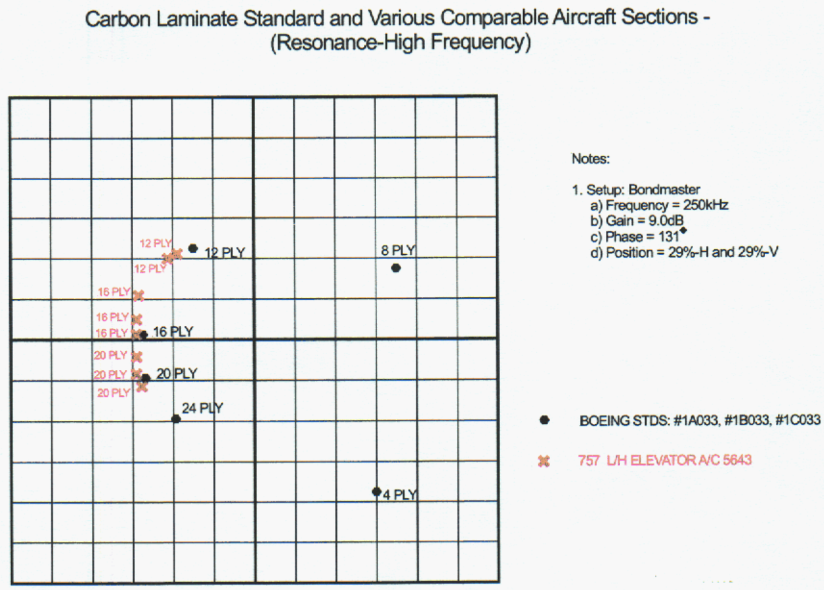


Figure 37: Laminate Response Variations Observed in the Field on a 757 L/H Elevator Structure – Aircraft #1

Carbon Laminate Standard and Various Comparable Aircraft Sections -
(Resonance-High Frequency)

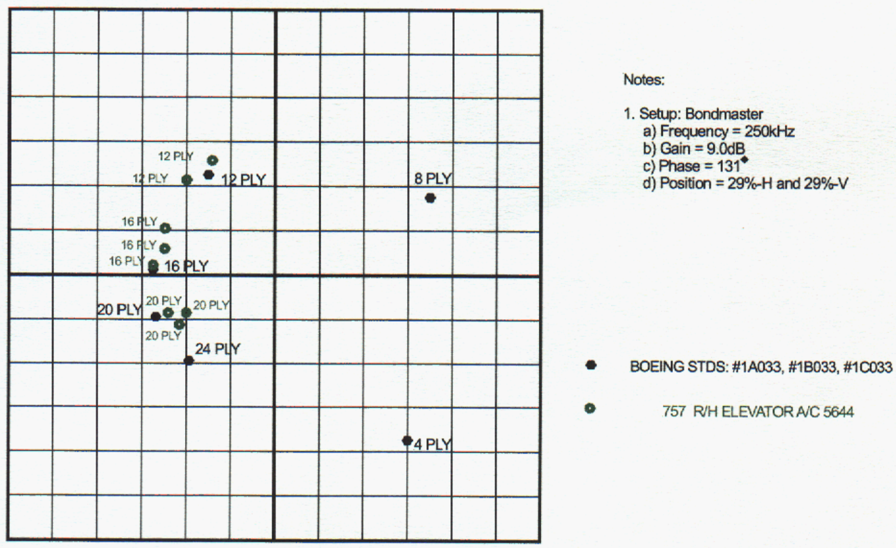


Figure 38: Laminate Response Variations Observed in the Field on a 757 R/H Elevator Structure – Aircraft #2

Carbon Laminate Standard and Various Comparable Aircraft Sections -
(Resonance-High Frequency)

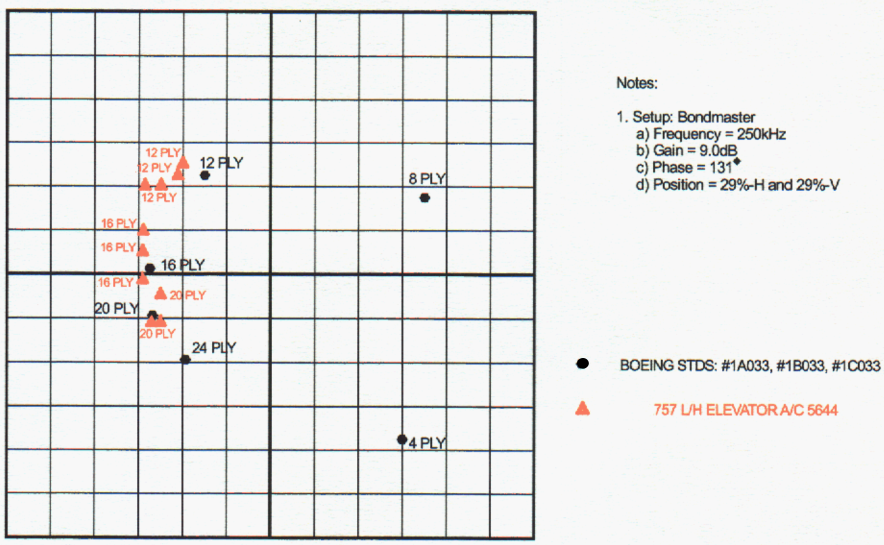


Figure 39: Laminate Response Variations Observed in the Field on a 757 L/H Elevator Structure – Aircraft #2

10.4 QUANTITATIVE THICKNESS MEASUREMENT TESTS

Validation tests were performed using the generic G11 standard to support inspections on a number of solid laminate aircraft structures with known flaws. A series of laminate thickness and delamination depth measurements were made using the G11 response curve on the Bondmaster device. These results were compared with: 1) thickness/depth predictions produced by existing laminate standards (resonance mode), and 2) thickness/depth predictions using pulse-echo (after calibrating velocities on existing laminate standards). The results, shown in Table 5, revealed that the G11 material is able to match the thickness/depth predictions to within approximately one ply of material (0.008" to 0.010" thick). This should provide acceptable thickness/depth predictions to support aircraft laminate inspections.

Table 5: Comparison of Thickness and Delamination Depth Measurements Made by Different NDI Techniques

Part	G11 Indication	Indication from Existing Laminate Standard	Pulse-Echo Reading
Carbon Honeycomb Std. with 6 Ply Laminate (delamination flaw)	0.030"	0.036"	0.037"
Carbon Honeycomb Std. with 6 Ply Laminate (disbond flaw)	0.047"	0.053"	0.057"
Fiberglass Honeycomb Std. with 6 Ply Laminate – Boeing Repair Std. (flaw #1)	0.040"	0.040"	0.048"
Fiberglass Honeycomb Std. with 6 Ply Laminate – Boeing Repair Std. (flaw #2)	0.050"	0.050"	0.058"
757 Elevator Panel #1 (laminate thickness)	0.093"	Not Measured	0.096"
757 Elevator Panel #1 (delamination thickness)	0.032"	Not Measured	0.038" *
757 Elevator Panel #1 (laminate thickness)	0.170"	Not Measured	0.167"
757 Elevator Panel #1 (delamination thickness)	0.054"	Not Measured	0.068" *
757 Elevator Panel #2 (laminate thickness)	0.054"	Not Measured	0.065"
757 Elevator Panel #1 (delamination thickness)	0.036"	Not Measured	0.046" *

* *Pulse-echo readings may not be accurate since flaws were induced by lightning strike and material density may have been altered. Thus, velocity calibrations on carbon laminate standards may not be accurate for determining depth of flaws.*

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11. VALIDATION TESTING USING AUTOMATED ULTRASONIC C-SCAN TECHNIQUES

11.1 GOAL OF FOCUSED TESTING WITH MAUS

The goal of this focused testing with the Mobile Automated Scanner (MAUS) NDI device was to acquire data that compliments the spiral response curves acquired with the Bondmaster device used in resonance mode. In order to eliminate data variations stemming from changes in probe deployment during hand-held testing, a series of laminate specimens were inspected side-by-side using the MAUS NDI system in resonance mode (C-scan data acquisition method). The MAUS inspection device, shown in Figure 40, provides for automated X-Y scanning with consistent spring-loaded deployment of the transducer.

11.2 VALIDATION TESTING VIA MAUS IN RESONANCE MODE

For resonance testing a single resonance transducer rated at 100 kHz was tuned in air with a computer generated frequency of 109 kHz (optimal sensor frequency can be chosen for the laminate thickness of interest). The sensor was nulled over the laminate thickness of interest on the G11 material. Without renulling the instrument, a scan was made on all other laminate standard panels that contained a matching thickness to obtain the image and comparative raw data counts of the different materials. Each scan also contained a thickness greater than and less than the area of interest to show the variations in color/response corresponding to slight changes in material thickness (see also MAUS C-scan test set-up in Appendix C).

The test specimens included existing fiberglass and carbon laminate standards, as well as, the candidate G11 material and various step wedges manufactured to simulate laminate aircraft structures. Existing fiberglass and carbon step wedge reference standards, along with other carbon laminates representing fabrication variability, were inspected along with the candidate G11 prototype standard (see Solid Laminate Test Specimen Matrix in Appendix C). The color coded images provided by the MAUS system can quantitatively show the similarity or difference in NDI responses obtained from similar thicknesses on all of the specimens. Similar responses for common thicknesses, indicated by the same color in the C-scan images, provides further evidence that the G11 material can be used as an NDI reference standard to support inspections of both fiberglass and carbon solid laminate structures.

<u>MAUS C-Scan Test Results</u> - The results, shown in Figures 41 - 46, show good agreement between the G11 phenolic and comparable thicknesses in the other specimens. The circled regions on each C-scan in the figures highlight the similar thickness regions for comparison. Figures 41-43 compare the G11 standard with a wide range of other industry laminate standards, including different materials and different composite weaves. The color codes show the strength of resonance response and indicate a good match between G11 and the other standards. Figures 44-46 compare the G11 standard with resonance scans from similar structures prepared using slightly different manufacturing processes. The comparisons show possible response variations stemming from aircraft structure with representative fabrication variability. These results provide some insight into the response variations that can exist in similar laminates. They demonstrate that even though composite aircraft structures may be similar, they do not display the NDI response consistency seen in metallic structures. There will be some envelope of

acceptable response signals. This is an important consideration when determining how closely the G11 response must match the existing laminate standards.

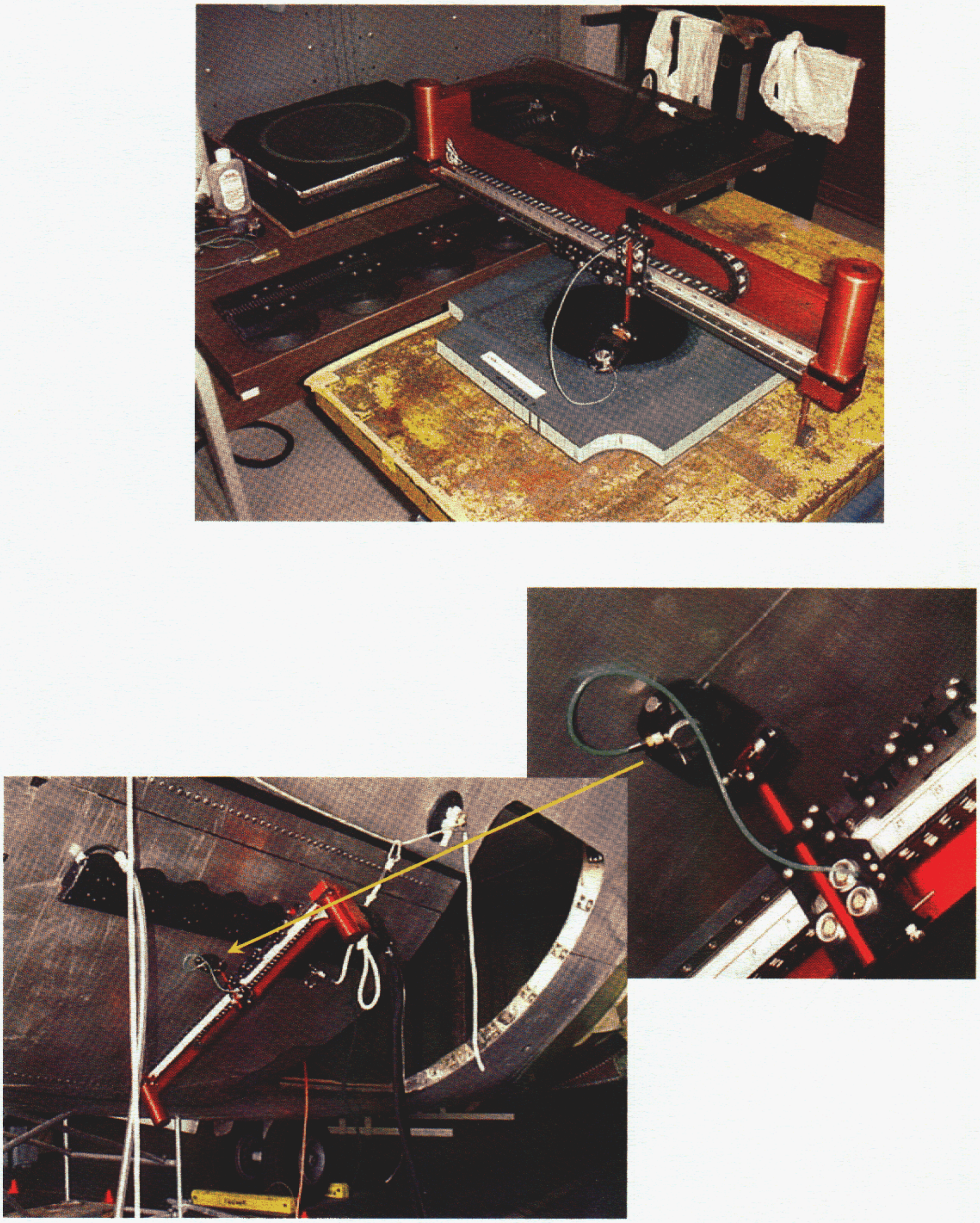

Figure 40: MAUS Inspections on Composite Honeycomb Repair Panel and Aircraft Fuselage Section

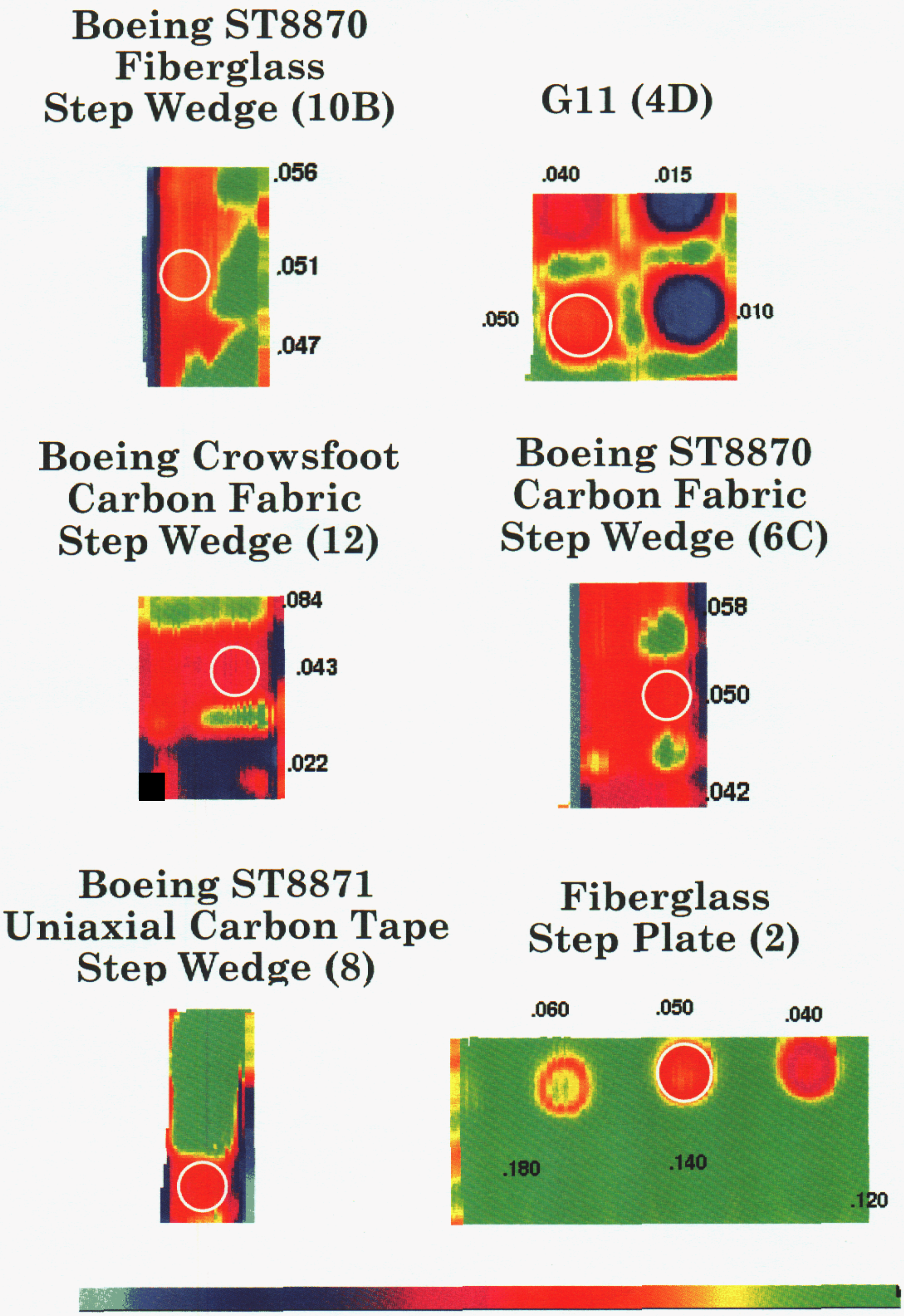

Figure 41: Comparison of 0.050" Thick Laminate Steps for G11 and Existing Industry Standards

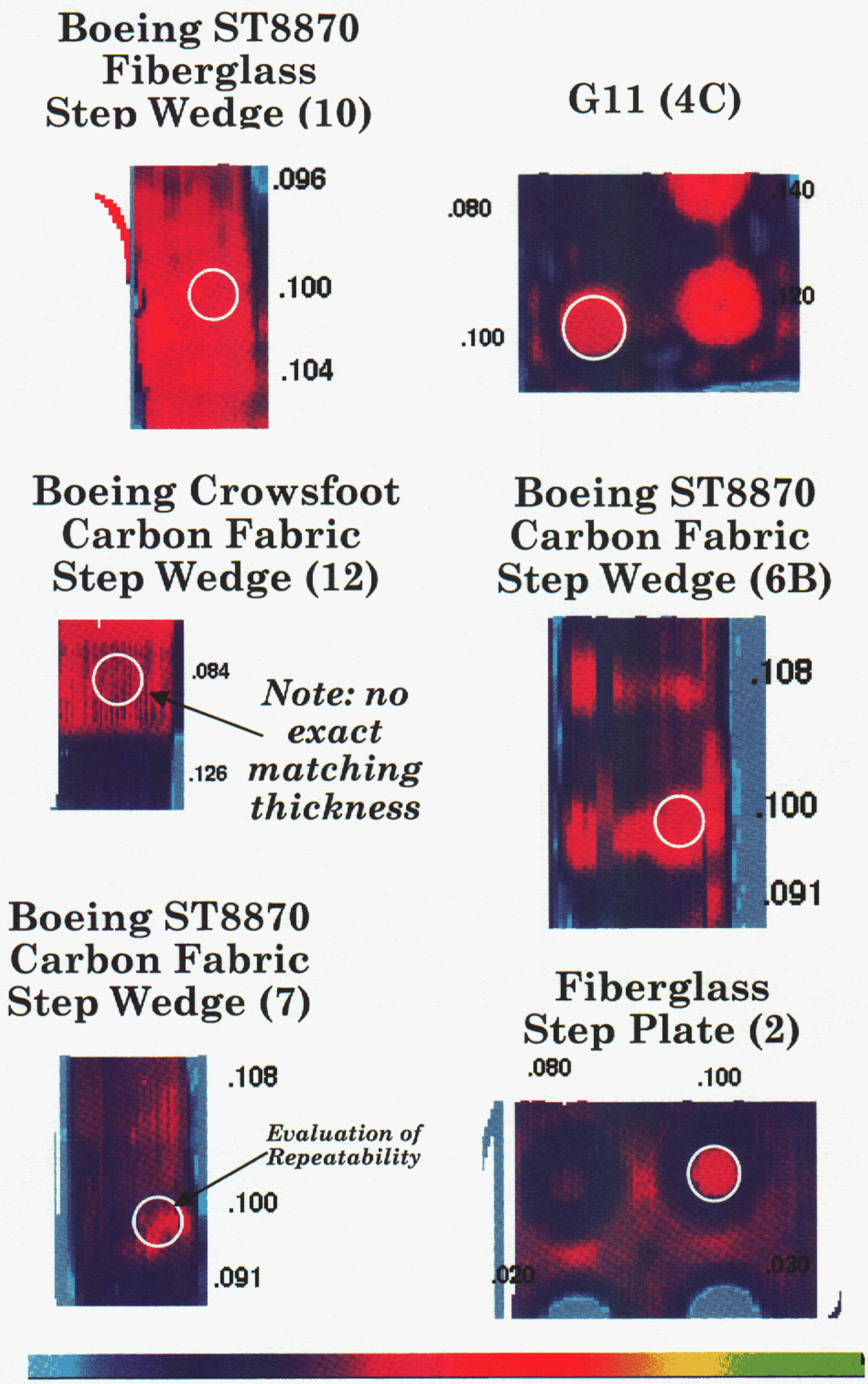

Figure 42: Comparison of 0.100" Thick Laminate Steps for G11 and Existing Industry Standards

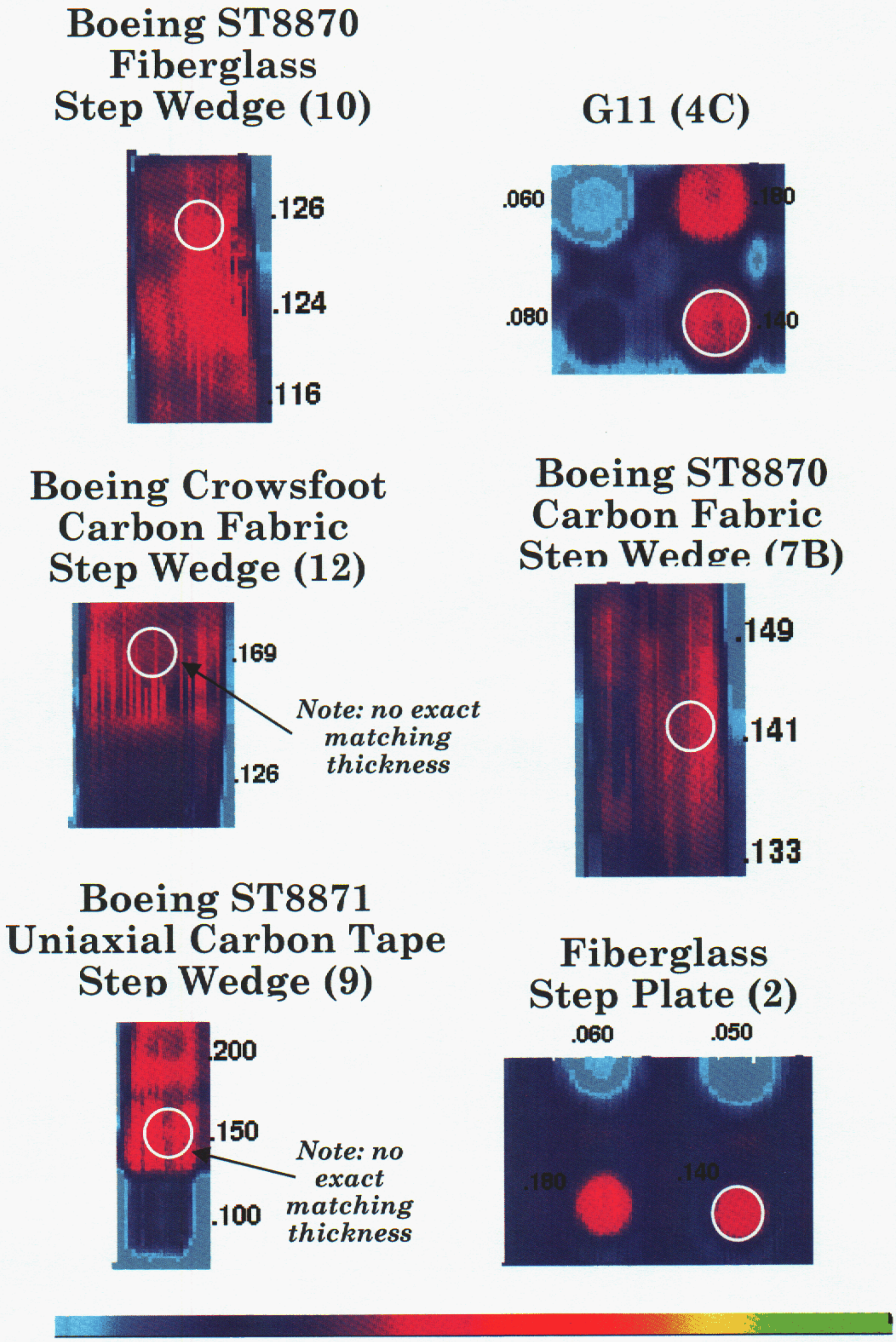

Figure 43: Comparison of 0.140" Thick Laminate Steps for G11 and Existing Industry Standards

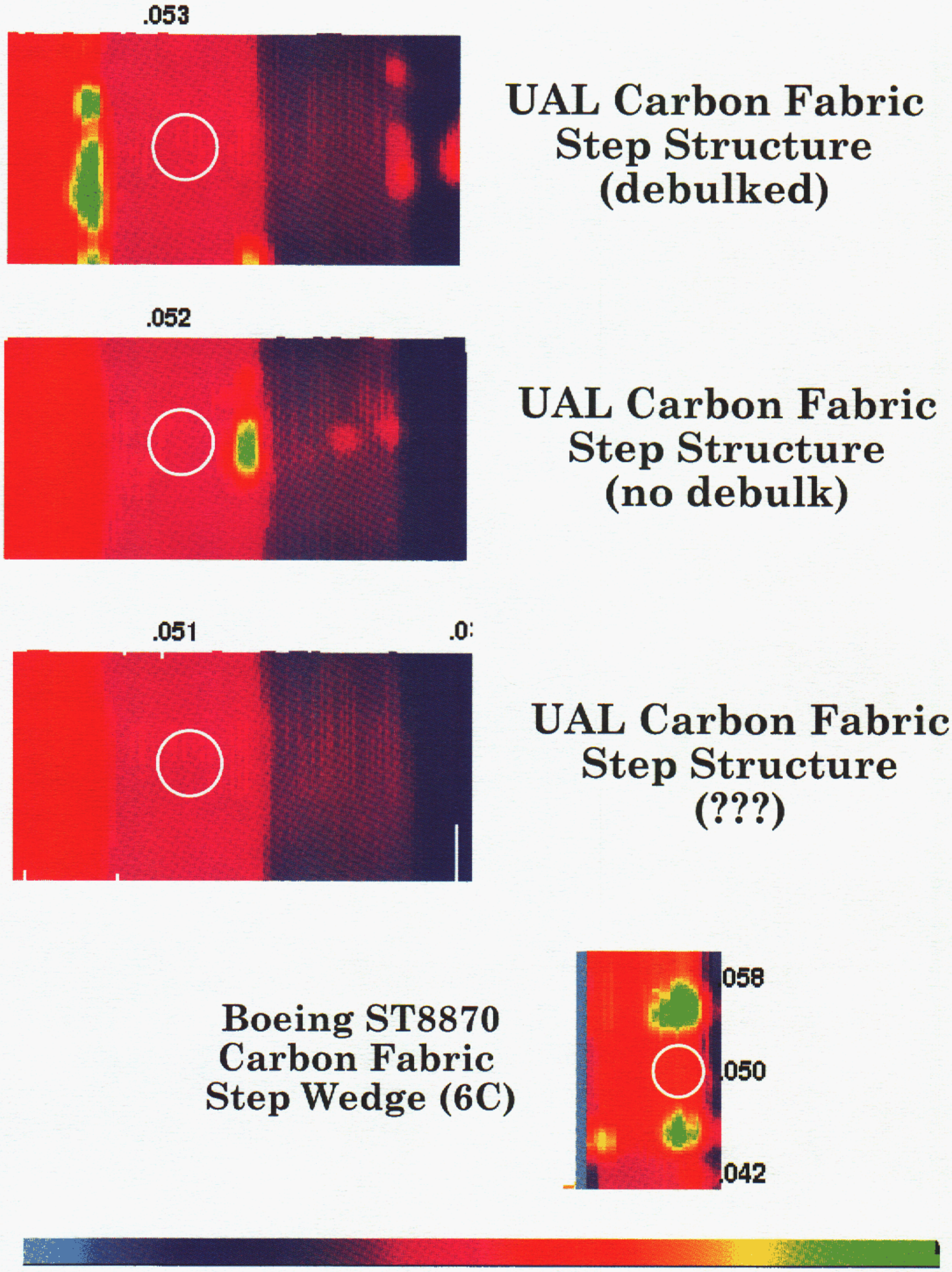

Figure 44: Comparison of 0.050" Thick Boeing Standard Step and Laminate Structures Fabricated in the Field

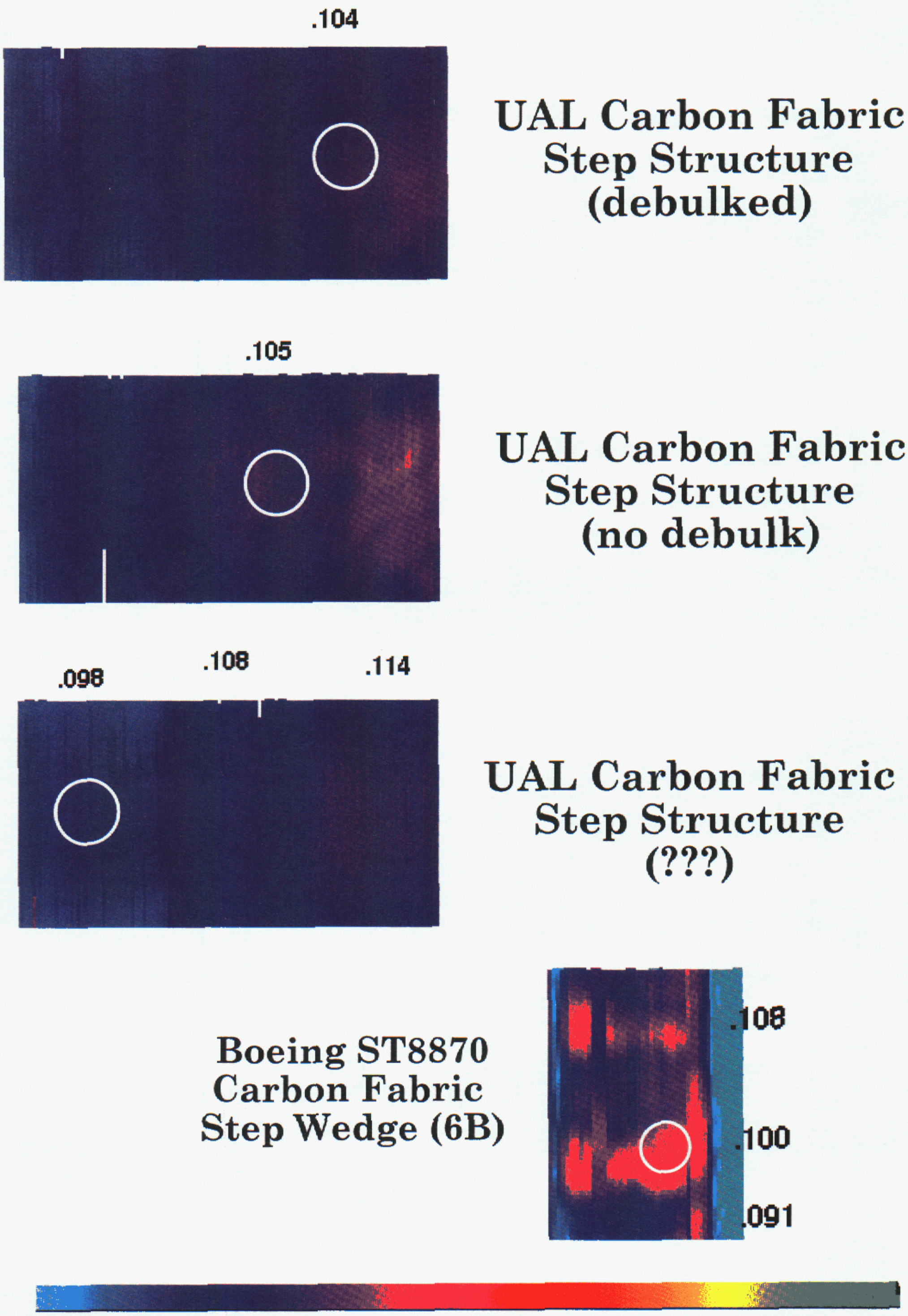

Figure 45: Comparison of 0.100" Thick Boeing Standard Step and Laminate Structures Fabricated in the Field

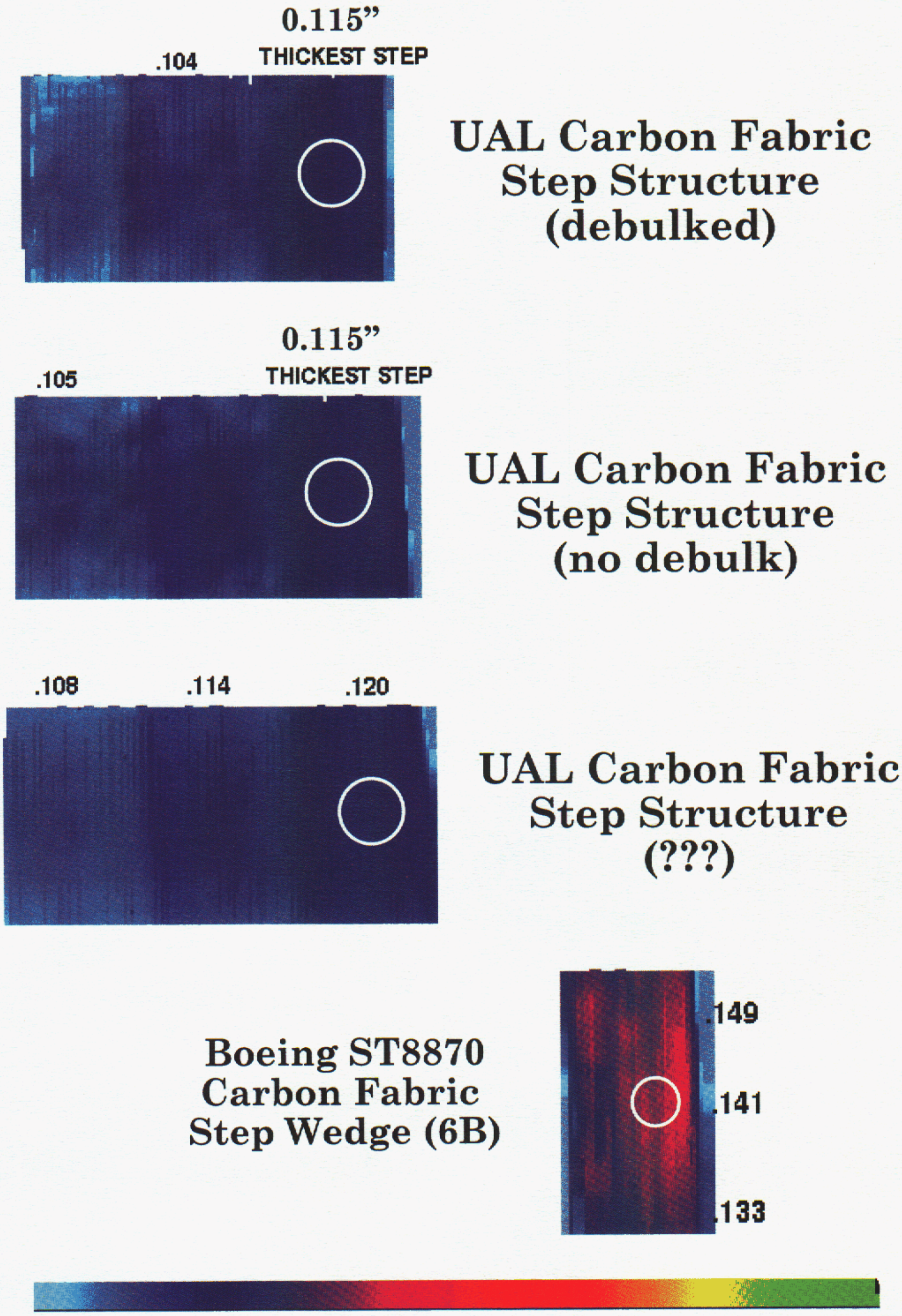

Figure 46: Comparison of 0.140" Thick Boeing Laminate Step and Laminate Structures Fabricated in the Field

12. ADOPTION OF LAMINATE STANDARDS BY AVIATION INDUSTRY

12.1 SAE AEROSPACE RECOMMENDED PRACTICE (ARP) AND MODIFICATIONS TO OEM MANUALS

All aspects of the solid laminate NDI reference standard design and production have been formally documented in SAE Aerospace Recommended Practice (ARP) 5605 (see Appendix B and Ref. 7). The ARP includes design drawings, fabrication specifications, certification requirements and quality assurance measures for the standards. This provides the central reference point to control changes and to assure that OEMs world wide have access to the latest information. Aircraft manufacturers and airlines will be notified of any modifications to the standards through a revised edition of this ARP. OEM Nondestructive Testing Manuals have been modified to include drawings of the standards and a reference to the SAE ARP. A reference to recommended fabrication shops is made to ensure that maintenance depots are provided with consistent and valid NDI standards. The NDI reference standards described here were delivered to the following airlines and OEMs to support their inspection of solid composite laminate structures:

1. Air Canada
2. Air France
3. Air New Zealand
4. Airbus OEM
5. Aloha Airlines
6. American Airlines
7. Bell Helicopter OEM
8. Boeing OEM
9. Bombardier OEM
10. British Airways
11. Delta Air Lines
12. Embraer OEM
13. Japan Airlines
14. Lufthansa
15. Northwest Airlines
16. Quantas Airlines
17. United Airlines
18. US Airways

12.2 USE OF SOLID LAMINATE NDI REFERENCE STANDARDS

The inclusion of the standards in NDT manuals is being accompanied by guidance on what techniques work well with the standards. It is important to note where a lack of flaw detection in the standards is associated with a limitation of the technique as opposed to a limitation of the standards.

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CONCLUSIONS

While seeking the optimum, yet minimum number, of composite honeycomb reference standards needed to conduct inspections on commercial aircraft structure, this study determined the honeycomb construction parameters that have a major effect on NDI. These results were used to produce a set of Honeycomb NDI Reference Standards that are applicable to most honeycomb aircraft structures produced by manufacturers worldwide. The reference standard set successfully completed an in-depth NDI validation phase conducted by OEMs, airlines, and the AANC.

An extensive material search, accompanied by key NDI response studies, has produced a generic material for solid composite laminate standards that will accommodate inspections on the full array of fiberglass and carbon laminates found on aircraft. NDI responses match existing laminates and appear to be within structural variations found in the field and/or within one ply of actual depth. A set of Solid Laminate NDI Reference Standards, made from G11 Phenolic material, was demonstrated to provide the same NDI response as existing carbon and fiberglass standards. In addition, the G11 material improves on existing solid laminate standards because it is inexpensive, can be reliably manufactured and is easy to machine into a solid laminate standard (i.e. plate with multiple thicknesses). NDI validation of this material consisted of both pulse-echo (velocity based) and resonance mode (acoustic impedance based) inspections carried out in laboratory and field environments.

The primary deliverables from this NDI reference standard development effort include: 1) an optimum and minimum set of NDI honeycomb and solid laminate reference standards for accomplishment of damage assessment and post-repair inspection of all commercial aircraft composites, 2) a series of flawed composite honeycomb test specimens that isolate fabrication variables and bound the primary aircraft inspection demands for honeycomb structures, 3) the engineering justification and recommendations for minimizing the number of calibration standards needed to carry out composite inspections on aircraft, 4) field evaluations that successfully demonstrated use of the standards in aircraft maintenance depots, 5) formal documentation of the honeycomb and solid laminate standards in the form of Aerospace Recommended Practices ARP5605 and ARP5606, and 6) information regarding applications and limitations of NDI techniques which will aid NDI utilization efforts. The primary benefits to the aviation industry include: 1) a consistent approach to composite inspections, 2) a reduction in standard procurement costs, 3) a mechanism for assessing emerging NDI techniques, and 4) a general improvement in the inspection of composite structures.

Overall, this effort produced a uniform approach to the inspection of composite structures on aircraft. Following final validation, field testing, and design optimization on both solid laminate and honeycomb reference standards, formal modifications to appropriate OEM manuals were completed. Through the active participation of the OEM's, this project represents a harmonized approach by aircraft manufacturers worldwide. The end result will be more streamlined inspection set-ups for aircraft maintenance depots and improved inspections through the use of optimized NDI reference standards.

REFERENCES

1. "Composite Repair NDT/NDI Handbook," SAE Aerospace Recommended Practice ARP5089, November 1996, prepared by the ATA/IATA/SAE Commercial Aircraft Composite Repair Committee, Inspection Task Group.

2. Roach, D.P., Dorrell, L.R., Kollgaard, J., Dreher, T., "Improving Aircraft Composite Inspections Using Optimized Reference Standards", SAE Airframe Maintenance and Repair Conference, Nov. 1998, SAE Technical Paper 98AEMR-34.

3. Roach, D.P., Rackow, K.A., "Detection of Hidden Flaws in Aircraft Composite Honeycomb Structure: Experiment Protocols," FAA/AANC Structured NDI Validation Experiment document, January 2001.

4. Roach, D.P., Rackow, K.A., "Improving In-Service Inspection of Composite Structures: It's a Game of CATT and MAUS", FAA/NASA/DoD Aging Aircraft Conference, September 2003.

5. Roach, D.P., "An Inspector Calls - The Search for Hidden Flaws in Composite Honeycomb Structures," *Aerospace Testing International*, February 2004.

6. "Composite Honeycomb NDI Reference Standards," SAE Aerospace Recommended Practice ARP5606, September 2001, prepared by the ATA/IATA/SAE Commercial Aircraft Composite Repair Committee, Inspection Task Group.

7. "Solid Composite Laminate NDI Reference Standards," SAE Aerospace Recommended Practice ARP5605, September 2001. prepared by the ATA/IATA/SAE Commercial Aircraft Composite Repair Committee, Inspection Task Group.

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APPENDIX A

Aerospace Recommended Practice 5606: Composite Honeycomb NDI Reference Standards

Aerospace Recommended Practice

Composite Honeycomb NDI Reference Standards

TABLE OF CONTENTS

Nomenclature:

ARP	Aerospace Recommended Practice
NDI	Nondestructive Inspection
NDT	Nondestructive Testing
NEMA	National Electric Manufacturers Association
OEM	Original Equipment Manufacturer

1. SCOPE

This recommended practice establishes generic reference standards that will accommodate nondestructive inspections (NDI) on a broad range of non-metallic composite honeycomb structures found on aircraft.

1.1 Purpose

The purpose of this Aerospace Recommended Practice is to describe the design and production of composite honeycomb calibration standards to be used in ultrasonic, resonant, and tap test NDI equipment calibration for accomplishment of damage assessment and post-repair inspection of aircraft composites. These standards may also be appropriate for other NDT methods but will need to be assessed as appropriate prior to their use. The standards are representative of damage found in the field and include typical flaw scenarios such as disbonds and delaminations. It is intended that these standards be adopted by aircraft Original Equipment Manufacturers within procedures contained in their Nondestructive Testing Manuals. Depending on the nature of the inspection, it may be necessary to compensate for variations in material properties through the use of correction factors or by adjusting for these differences on the part or structure being inspected. In certain instances, it may be desirable or necessary to design a new reference standard to accommodate a specific inspection application.

Currently, the recognized number of composite honeycomb construction variables makes the resulting number of specimens very large and unmanageable. Inspection characterizations and equipment responses have been used to determine the important variables needed in a composite reference standard thus eliminating unnecessary standard configurations. This Aerospace Recommended Practice (ARP) describes a workable number of reference specimens that can meet the needs of a broad range of honeycomb structures found on aircraft.

1.2 Background

The CACRC Inspection Task Group developed this ARP in an effort to establish a single, generic set of composite honeycomb reference standards that would accommodate inspections on the majority of non-metallic honeycomb structures found on aircraft. The advantages of industry-wide accepted composite standards include: 1) providing a consistent approach to composite inspection thus helping to minimize false calls, 2) reducing standard procurement costs, and 3) aiding the assessment of composite inspection technologies. The goal of this project is to develop standards that will allow for repeatable, accurate inspections in light of increases in the number of composite structure inspection tasks. Specific use of the honeycomb standards described in this ARP can be achieved through the OEM inspection procedures found in Nondestructive Testing Manuals and Nondestructive Testing Standard Practice Manuals.

1.3 Supporting Data

The number of construction variables encountered in composite honeycomb structure makes the number of potential reference standards needed to support the inspections very large. In an effort to reduce the number of standards needed with proper engineering justification, key construction variables were identified and their affects on inspection results were assessed. The variables evaluated were: skin material, skin thickness, core material, core thickness, core weight, and cell size. Additionally, various methods of manufacturing flaws were evaluated to ensure repeatable and accurate representation of disbonds and delaminations. A suite of 64 honeycomb panels, representing reasonable bounding conditions of the construction variables listed above were manufactured and inspected using a wide array of sonic and ultrasonic NDI techniques. In this manner, the effects of each variable on NDI could be assessed in order to provide justification for minimizing the number of calibration standards.

An analysis of the resulting data identified skin material, skin thickness, and core material as the key variables affecting the inspection method used. A final set of minimum honeycomb reference standards were designed and fabricated to include these key variables. A sequence of NDI experiments were completed to demonstrate that this minimum honeycomb reference standard set is able to fully support inspections over a wide range of honeycomb construction scenarios.

1.4 Use of Standards

It is hoped that these honeycomb standards will be applicable to most composite honeycomb structures found on aircraft, however, the specific range of construction variables certified in this study are listed in Table 1. Reference [1] presents the NDI testing and analysis that was carried out to arrive at the final set of honeycomb standards described in this ARP. Specific testing will be needed to certify the use of these standards outside the type and range of variables listed in Table 1.

Furthermore, when using these standards, consideration needs to be given to surface coatings such as paint or lightning protection plies. This is a reference standard construction document and not an inspection document. Inspection procedures, from OEM or users' maintenance manuals, must accompany the use of these reference standards for each unique family of composite honeycomb construction.

Table 1: Range of Composite Honeycomb Construction Variables Tested to Arrive at the Standards Listed in this ARP

Composite Construction Variable	Bounding Conditions Studied
Laminate Material	Carbon & Fiberglass
Laminate Thickness	3 plies - 12 plies
Honeycomb Core Material	Nomex & Fiberglass
Honeycomb Core Thickness	0.25" - 2.0" thick (6.35 - 49 mm)
Honeycomb Cell Size	0.125" - 0.25" width (3.18 - 6.35 mm)
Honeycomb Core Density	2 - 8 lb/ft^3

2. APPLICABLE DOCUMENTS

The following publications form a part of this specification to the extent specified herein. The applicable issue of the referenced publications shall be the issue in effect on the date of the purchase order.

2.1 **U. S. Government Publications:**

Available from DODSSP, Subscription Services Desk, Building 4D, 700 Robbins Avenue, Philadelphia, PA 19111-5094.

SAE-AMS-C-9084 Cloth, Glass, Finished, For Resin Laminates

2.2 **Other Publications:**

Industry specifications are listed for information in section 4.1.

3. **TECHNICAL REQUIREMENTS**

3.1 **Fabrication and Materials**

Fabrication of the honeycomb/composite panels (See attached engineering design drawings CHRS-1 and CHRS-2) consists of three tasks: 1) fabrication of the composite laminate plates for the top and bottom of the sandwich assembly, 2) preparation of the honeycomb core material, and 3) secondary bond of laminate to honeycomb core to produce the honeycomb panels.

Appendix A contains a series of photos showing the steps involved in the three major panel production activities: 1) laminate skin preparation, 2) honeycomb core preparation, and 3) honeycomb panel assembly. These photos should be referenced as the fabrication instructions below are carried out.

3.1.1 **Laminate:**

Prepare the laminates as per the specimen drawings CHRS-1 and CHRS-2 contained in these instructions. Also see section 3.1.4 regarding the engineered flaws. Place laminates on smooth tool as per lay-up shown in Figure 1 and cure per the laminate temperature profiles shown in Figure 2. All "pillow insert" flaws are located two plies down from the "peel ply" side shown in Figure 1 below.

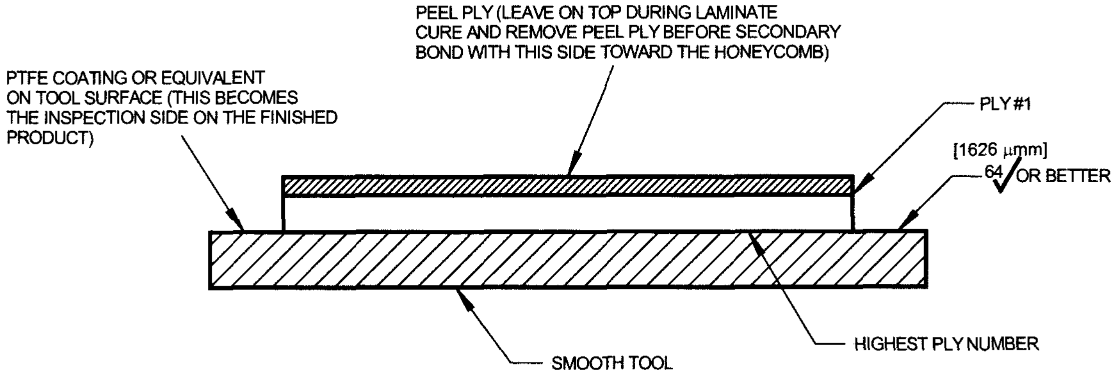


Figure 1: Laminate Lay-Up on Smooth Tool

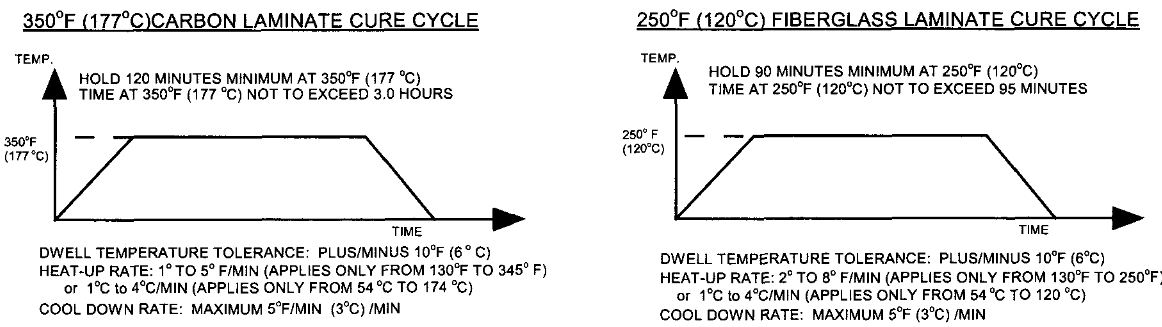


Figure 2: Laminate Cure Temperature Profile

Produce 16 laminate plates, two each of each material type (carbon and fiberglass) at 3, 6, 9, and 12 ply thicknesses. One laminate of each material type and ply thickness will contain flaws and the other will not. Size and layout for laminates during cure cycle is shown in Figure 3. Use an autoclave or automated oven and cure laminates in a vacuum bag at 11-12 psi (568.9 - 620.6 mm Hg; 75.9 - 82.7 kPa). *Note: In order to create the proper attenuation and desired laminate response properties do not exceed 12 psi in the cure pressure.*

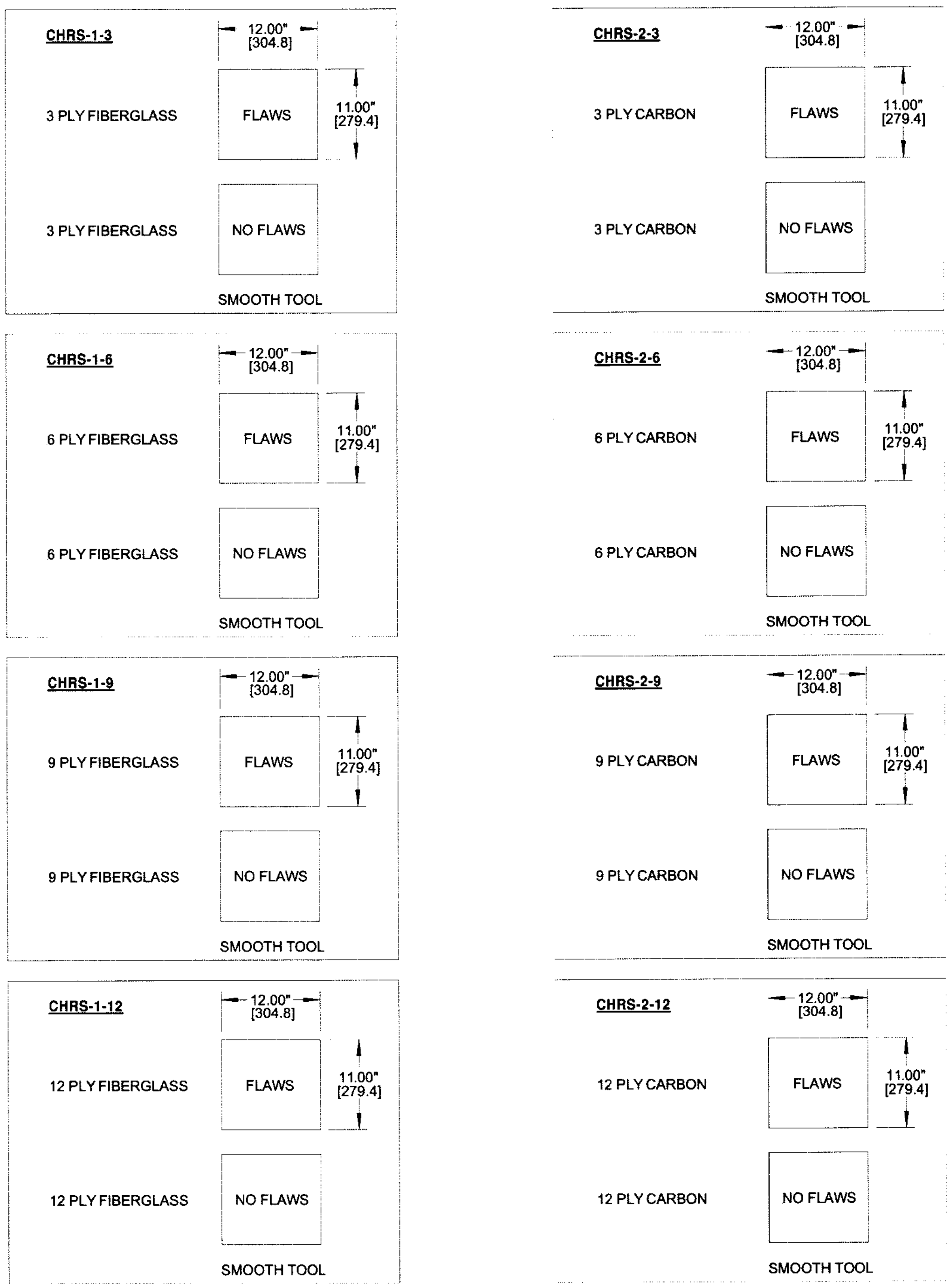


Figure 3: Summary of Individual Laminate Sheets
[Dimensions in brackets are in mm]

Note: There are two laminate plates per specimen – one with flaws and one without flaws. Laminates are cured first and then bonded to the honeycomb in a secondary process.

3.1.1.1 **LAMINATE LAY-UP:**

3 ply lay-up: [+45, 90, -45]
6 ply lay-up: $[+45, 90, -45]_2$
9 ply lay-up: $[+45, 90, -45]_3$
12 ply lay-up: $[+45, 90, -45]_4$

3.1.2 **HONEYCOMB**:

Prepare the honeycomb in accordance with the engineering drawings shown in Section 3.3 and summarized in Table 2. The honeycomb preparation consists of: 1) adding the machined core flaws, 2) producing a potted core region, and 3) making a core splice using a foaming adhesive. Before joining the laminates to the honeycomb ensure that the ribbon direction is on the X-axis along the 12.00 inch (304.8mm) dimension (See Figure 4). Join the laminates to the honeycomb using a secondary bond per the cure temperature profiles shown in Figure 5. The "tool side" of the laminate (see Fig. 1) should face outward (inspection surface) and the "peel ply" side of the laminate should be placed toward the honeycomb (bonded surface). For typical setup of honeycomb to laminate bond see Figure 6. *Note: actual peel ply should be removed from laminates before secondary bonding process.*

Table 2: Summary of Laminate and Honeycomb Types with Reference to Engineering Drawings for Fabrication

Engineering Drawing & Specimen Number	**Laminate Material**	***Core Material***	***Number of Plies***
CHRS-1-3	Fiberglass	Nomex & Fiberglass	3
CHRS-1-6	Fiberglass	Nomex & Fiberglass	6
CHRS-1-9	Fiberglass	Nomex & Fiberglass	9
CHRS-1-12	Fiberglass	Nomex & Fiberglass	12
CHRS-2-3	Carbon Fabric	Nomex & Fiberglass	3
CHRS-2-6	Carbon Fabric	Nomex & Fiberglass	6
CHRS-2-9	Carbon Fabric	Nomex & Fiberglass	9
CHRS-2-12	Carbon Fabric	Nomex & Fiberglass	12

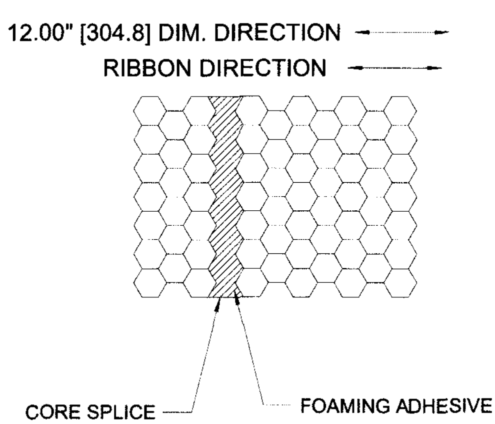


Figure 4: Honeycomb Ribbon Direction

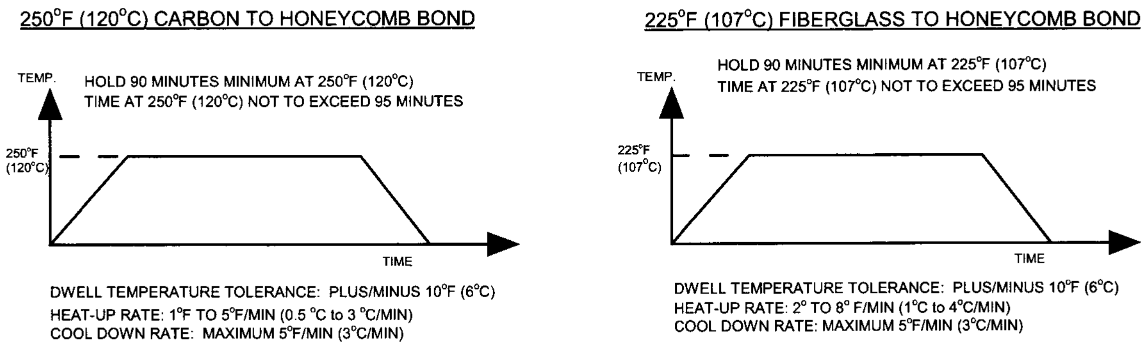


Figure 5: Cure Temperature Profile for Secondary Bond of Laminate to Honeycomb Core

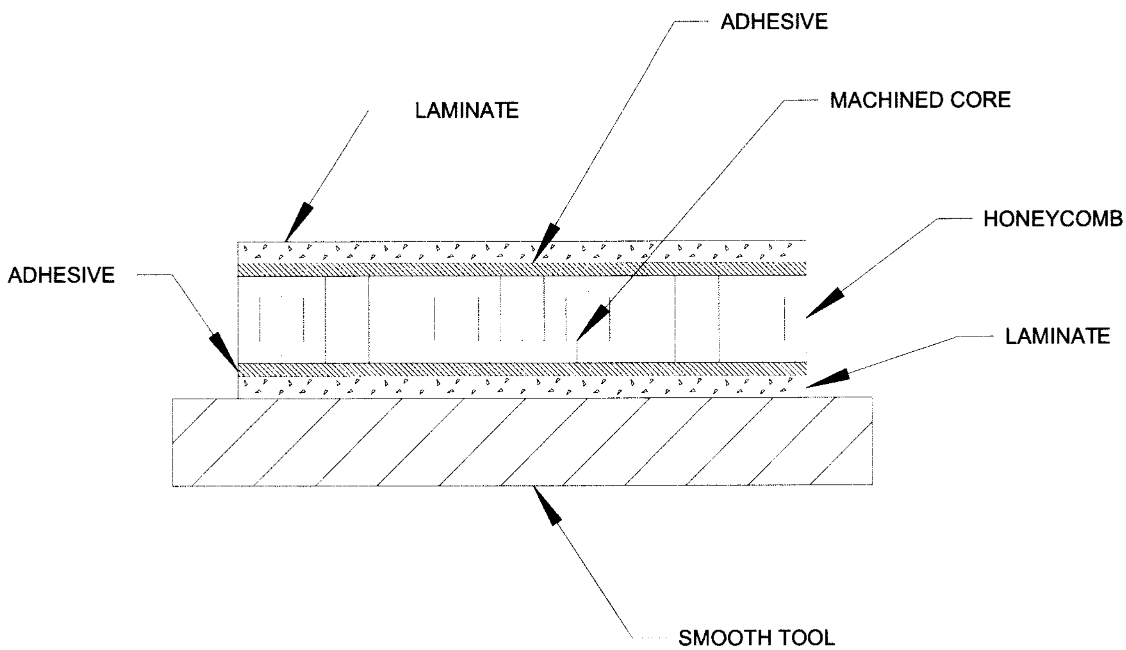


Figure 6: Laminate to Honeycomb Lay-up

3.1.3 Cure Pressure

Use of an autoclave or an automated oven will be required for all bonds. Cure all bonds in a vacuum bag at 11-12 psi (568.9 - 620.6 mm Hg ; 75.9 - 82.7 kPa).

3.1.4 Engineered Flaws and Special Panel Areas

a) Skin-to-Honeycomb Disbond

Machined core areas will be milled out of the honeycomb using a dremel tool, router, or equivalent to produce a flat-bottomed hole as per Figure 6 (also see Appendix A). Depth of machined core flaws in the honeycomb will be approximately 0.250 inch (6.35mm).

b) Pillow Inserts

Pillow inserts consist of four layers of tissue held together between two layers of heat resistant, polyamide film tape (See Figure 7). Insert the 1.0 inch (25.4mm) diameter pillow inserts into the laminate lay-up at the locations called out in the specimen drawings CHRS-1 and CHRS-2.

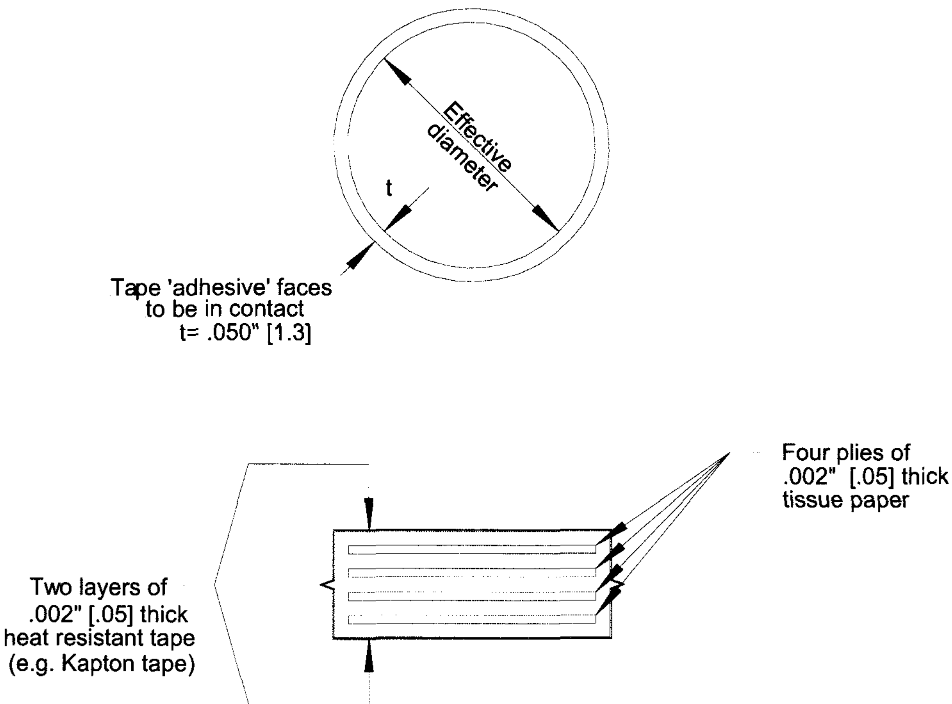


Figure 7: Pillow Insert Construction

c) Core Splice Area

Core splice area will be fabricated using a single strip of foaming adhesive. Note the ribbon direction as shown in Figure 4. Cure under a vacuum bag at 250°F (120°C) for 90 minutes.

d) Potted Core Areas

Potted core areas will consist of filling the honeycomb cells in a 1.0 inch (25.4mm) diameter area with potting material and curing under a vacuum bag for 90 minutes at 250°F (120°C). The individual cells can be filled or all cells in the 1.0 inch (25.4mm) diameter region can be removed and the entire area potted with one fill. The summary of the process is as follows. Place masking tape over the bottom of the cells that are to be filled with potting material (See Figure 8). Mix the core potting material thoroughly in a container. Draw a vacuum on the container to remove any trapped air in the mixture. Use a syringe to inject the potting material into each cell within the 1 inch (25.4mm) diameter area of interest (See Figure 8). Make sure that the potting material is flush with the top of each cell. After the material has cured sand off any excess so that the potting material is not above the top of the cells. Step-by-step directions for producing potted cores follows.

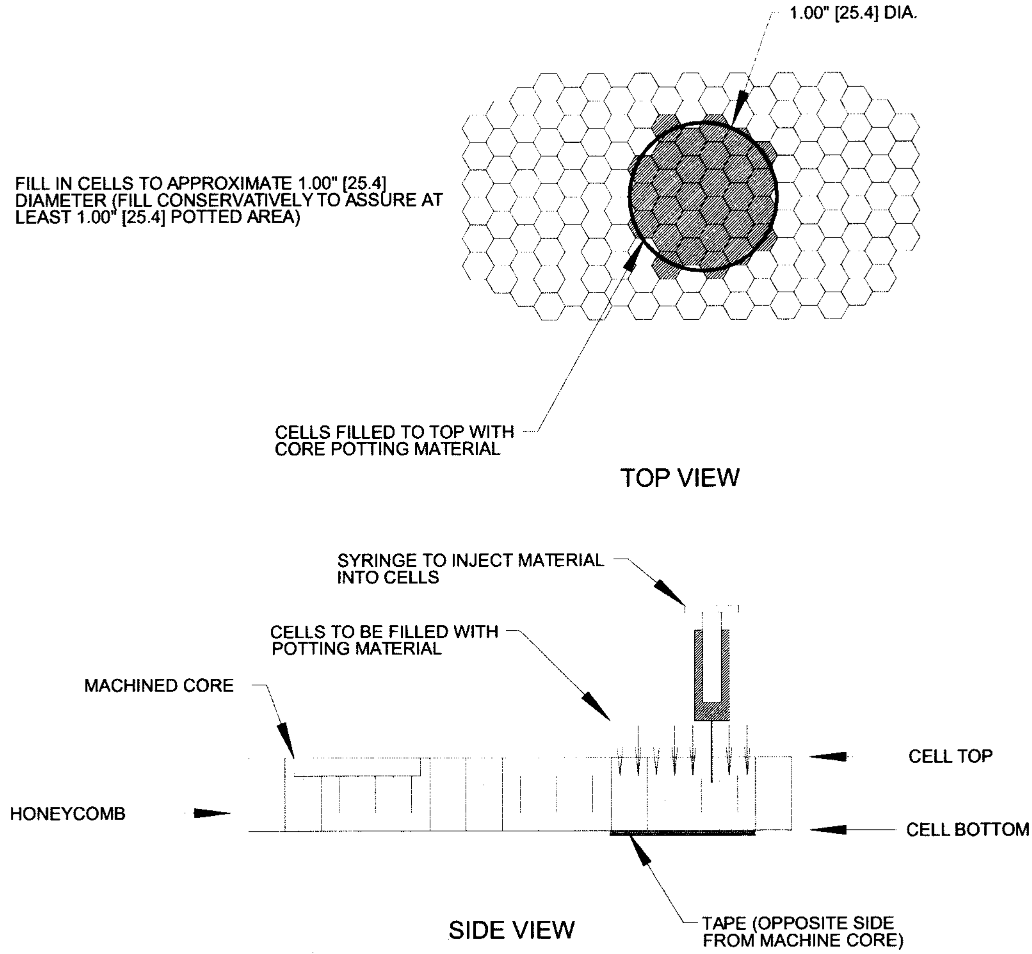


Figure 8: Process for Creating Potted Honeycomb Core Areas

3.1.4.1 Step-By-Step Directions for Producing Potted Honeycomb Core Regions

Figure 8 and Appendix A should be referenced while following these directions for producing potted cores.

a) Prepare a half-batch of CG1305 epoxy resin (or equivalent – see materials list 4.1). The CG1305 standard ratio is 100/20.

- Part A: 50 grams
- Part B: 10 grams

Note: With this batch size, two 1inch (25.4 mm) diameter sized areas can be encapsulated before the resin starts to gel; 30 minute pot life. A power syringe can be used if large areas need to be encapsulated. If the material has started to gel discontinue process and prepare another fresh batch of resin.

b) Mix batch by hand for 2 minutes using a wooden tongue depressor, spatula, or equivalent.

c) Place mixing container with material into an vacuum chamber and evacuate the mixture (approximately 10-11 psi) until all volume reduction is achieved. Remove mixing container from the vacuum chamber.

d) Pour the epoxy into a 30 cc plastic syringe. Set the syringe on its base and allow any air bubbles to rise to the free surface.

Note: While filling the syringe, tilt the syringe at an angle and slowly pour the material into the syringe. Avoid trapping or generating any air bubbles in the resin system.

e) Place a metallic syringe needle onto the syringe. Use a minimum 1/16 inch (1.6 mm) diameter ID needle with this resin system. The length of the needle will be determined by the thickness of the honeycomb cells being filled. The needle should reach to the base of the honeycomb. After the resin is void-free. Run the plunger up though the syringe to eliminate any free space in the syringe.

f) Seal the base of the honeycomb cells that are to be filled with potting material using masking tape. Make sure that the panels and work surfaces are flat and parallel. Assorted plates can be used as weights to ensure that the honeycomb panels are kept flat around the areas to be potted.

g) Carefully begin filling the honeycomb cells by inserting the needle into each cell. The needle should be touching the bottom of the cell. Slowly fill the cell approximately 3/4 full withdrawing the needle as the material fills the cell. Remove the syringe and continue filling the desired area/pattern in the honeycomb. After the area has been filled, top off any cells which need additional resin.

h) Cure for 2-3 hours at room temperature followed by a 250°F (120°C) post-cure per specifications (or other as per manufacturer's specifications).

Note: The masking tape should be removed before the panels are exposed to the 250°F (120°C) post-cure. Maintain flatness during the post-cure. Thin PTFE sheets, flat plates, and dead weights can be used to keep the panels flat. Make sure that any weighting system is distributed evenly over the honeycomb. Localized weights may crush honeycomb cells.

3.1.5 **Sealing**

When the laminates have been bonded to the honeycomb core, a complete composite/honeycomb sandwich assembly will be produced in accordance with Figure 9. It is now necessary to seal each panel around the periphery to avoid moisture ingress and to provide mechanical protection. The sealing process is as follows. Rout the honeycomb to remove approximately .250 inch (6.35mm) of honeycomb material around the perimeter of the panel. The honeycomb is now recessed from the upper and lower laminates as shown in Figure 9. Seal all honeycomb edges using the sealant called out in section 4.1 "Materials". It may be necessary to add a stiffener such as milled fiberglass to the sealant for easier workability and proper set-up. Once the sealant is set, it should be sanded/finished such that it is smooth and flush with the laminate edges (See Figure 9).

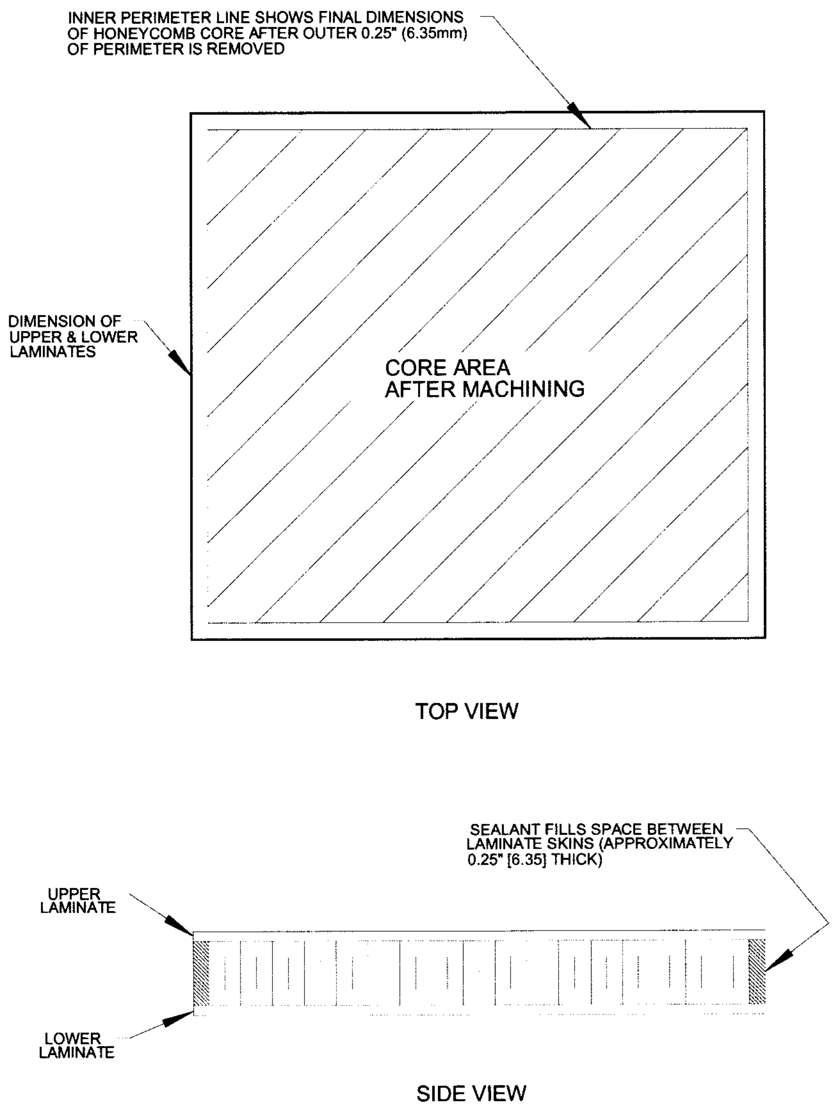


Figure 9: Application of Sealant Around Perimeter of Honeycomb Panels

3.1.6 Materials

Representative materials are listed in 4.1.

3.1.7 General Requirements

a) All other aspects of fabrication (surface preparation, clean room, etc.) should be in accordance with industry standards.

b) Fill in the checklist/as-built form to verify materials and processes used during the construction of the panels. Provide completed "As-Built" forms to customer.

c) Ensure that marking of specimen numbers and flaw locations is permanent. Label the flaw side of each finished product with the corresponding specimen number (see Drawings CHRS-1 and CHRS-2).

d) Numbers that are in [ ] on Figures are millimeters.

e) Perform a certification inspection in accordance with 3.2 "Acceptance Criteria." Provide C-scan results, with attenuation levels labeled as specified in section 3.2, to customer.

3.2 Acceptance Criteria

The acceptance criteria shall be as defined in Table 3 and Figure 10.

Table 3: Acceptance Criteria for Ultrasonic Inspection of Standards

Reference Standard Location	Acceptance Limits *
Pillow Insert (Interply Delamination)	The ultrasonic attenuation of the Pillow Insert areas must be at least 12dB greater than the attenuation of the Ref. Std. areas without defects.
Machined Core (Disbond)	The ultrasonic attenuation of the Machined Core areas must be at least 12dB greater than the attenuation of the Ref. Std. areas without defects.
Potted Core	The ultrasonic attenuation of the Potted Core areas must be at least 6dB less than the attenuation of the Ref. Std. areas without defects.
Unflawed Area	The ultrasonic attenuation of unflawed areas must be at least 18dB less than the attenuation of the foam tape on the Ref. Std.

**Use a 1 MHz Through-Transmission Ultrasonic (TTU) inspection system.*

3.2.1 The Reference Standards must be certified using a Through Transmission Ultrasonic (TTU) 'C' scan inspection. Label each flaw with the corresponding attenuation value determined by the TTU inspection.

3.2.2 Ultrasonic indications outside the defect areas must be no greater than 0.50 inch (12.7mm) in diameter. An ultrasonic indication is an area with ultrasonic attenuation that is at least 6 dB larger than the attenuation of the adjacent areas without defects. Multiple indications must be at least 1.00 inch (25.4 mm) apart. There should be no more than three (3) anomaly indications in the non-defect regions of the specimen. If there are more than three areas with deviations of 6 dB or more, the panel is rejected. The location of all of these UT indication regions should be permanently marked on the standard to show "no calibration" areas on the specimen. If any UT indication in the unflawed region exceeds 18 dB, the specimen shall be rejected.

3.2.3 See Figure 10 for Attenuation Acceptance Limits.

3.2.4 Flaw Sizing: A 1.0 inch [25.4 mm] diameter piece of foam tape (see 4.1 Materials) shall be placed on the specimen during the TTU inspection. In addition to providing relative attenuation levels for flaw certification, the tape will be used to ensure correct flaw sizing. Indications from the machined core and pillow insert flaws shall be recorded on the TTU C-scan image. The size of these manufactured flaws shall be within ± 10% of the foam tape anomaly size shown in the C-scan. As per Table 3, the manufactured flaw areas should produce 12 dB or larger attenuation. Individual, unflawed areas surrounding the manufactured flaws that produce less than 12 dB but greater than 3 dB of attenuation shall not exceed 20% of the flaw dimension.

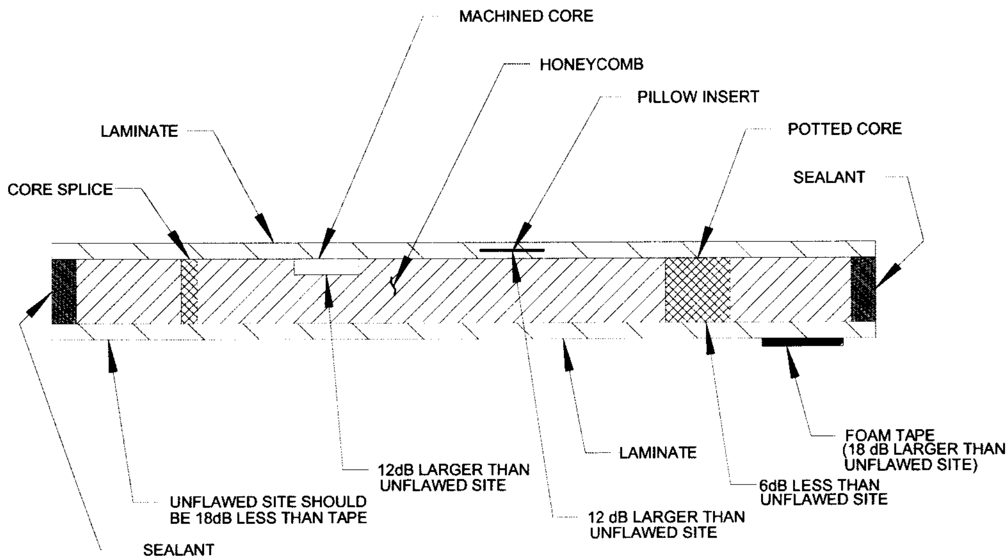
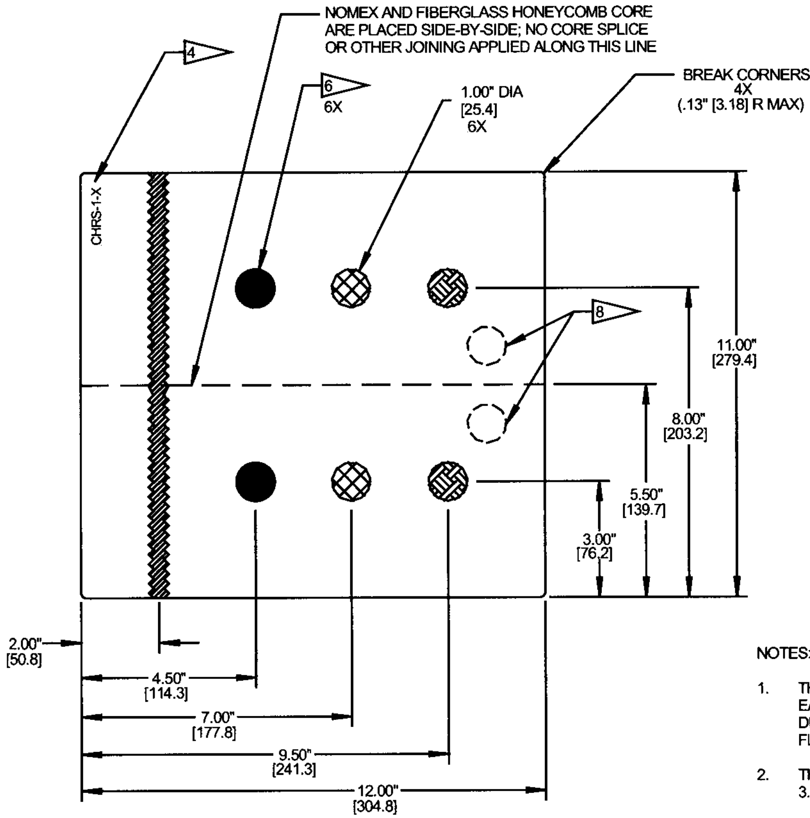
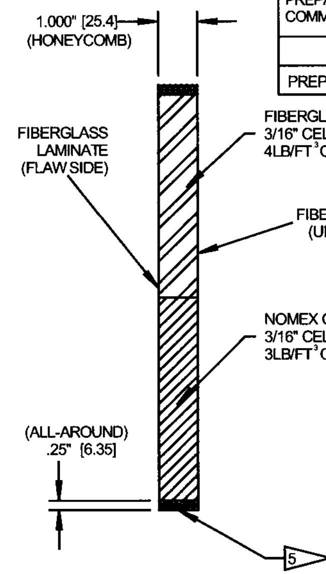


Figure 10: Attenuation Acceptance Limits

3.3 Engineering Design Drawings for Fiberglass Skin (CHRS-1) and Carbon Skin (CHRS-2) Honeycomb Reference Standards:

FIBERGLASS REFERENCE STANDARD		
DATE: 2-12-01	REV. #1	SCALE:1/2
PREPARED UNDER THE JURISDICTION OF ATA/IATA/SAE COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE		
DWG. NO. CHRS-1		
PREPARED BY: CACRC INSPECTION TASK GROUP		





MACHINED CORE (DEPTH OF 0.25" [6.35] IN HONEYCOMB)

PILLOW INSERTS (LOCATED TWO PLIES UP FROM CORE MATERIAL)

POTTED CORE

CORE SPLICE

NOTES:

1. THERE WILL BE FOUR FIBERGLASS SPECIMENS FABRICATED USING THIS DESIGN, EACH WITH DIFFERENT THICKNESS LAMINATES. THE NUMBER OF PLIES FOR THE DIFFERENT THICKNESSES WILL BE 3, 6, 9, AND 12. ONE SIDE WILL CONTAIN FLAW INSERTS AND THE OTHER SIDE WILL REMAIN UNFLAWED.

2. THE MACHINE CORE IS TO PRODUCE SKIN-TO-CORE DISBONDS, FABRICATE PER SECTION 3.1.4 OF THIS AEROSPACE RECOMMENDED PRACTICE.

3. PILLOW INSERTS ARE TO PRODUCE INTERPLY DELAMINATIONS, FABRICATE PER SECTION 3.1.4 OF THIS AEROSPACE RECOMMENDED PRACTICE.

4. MARK THE FIBERGLASS SKIN SPECIMEN NOS. USING A METHOD THAT IS PERMANENT. THE NOS. SHALL BE CHRS-1-3 FOR 3-PLY, CHRS-1-6 FOR 6-PLY, CHRS-1-9 FOR 9-PLY AND CHRS-1-12 FOR 12 PLY. LOCATE APPROXIMATELY AS SHOWN USING .25" [6.35] HIGH CHARACTERS.

5. NOTE AFTER CURING, THE EDGES OF HONEYCOMB ARE ROUTED TO MAKE DIMENSIONS LESS THAN THE LAMINATE (SKIN). THE EDGES ARE SEALED AS PER SECTION 3.1.5 IN THIS AEROSPACE RECOMMENDED PRACTICE.

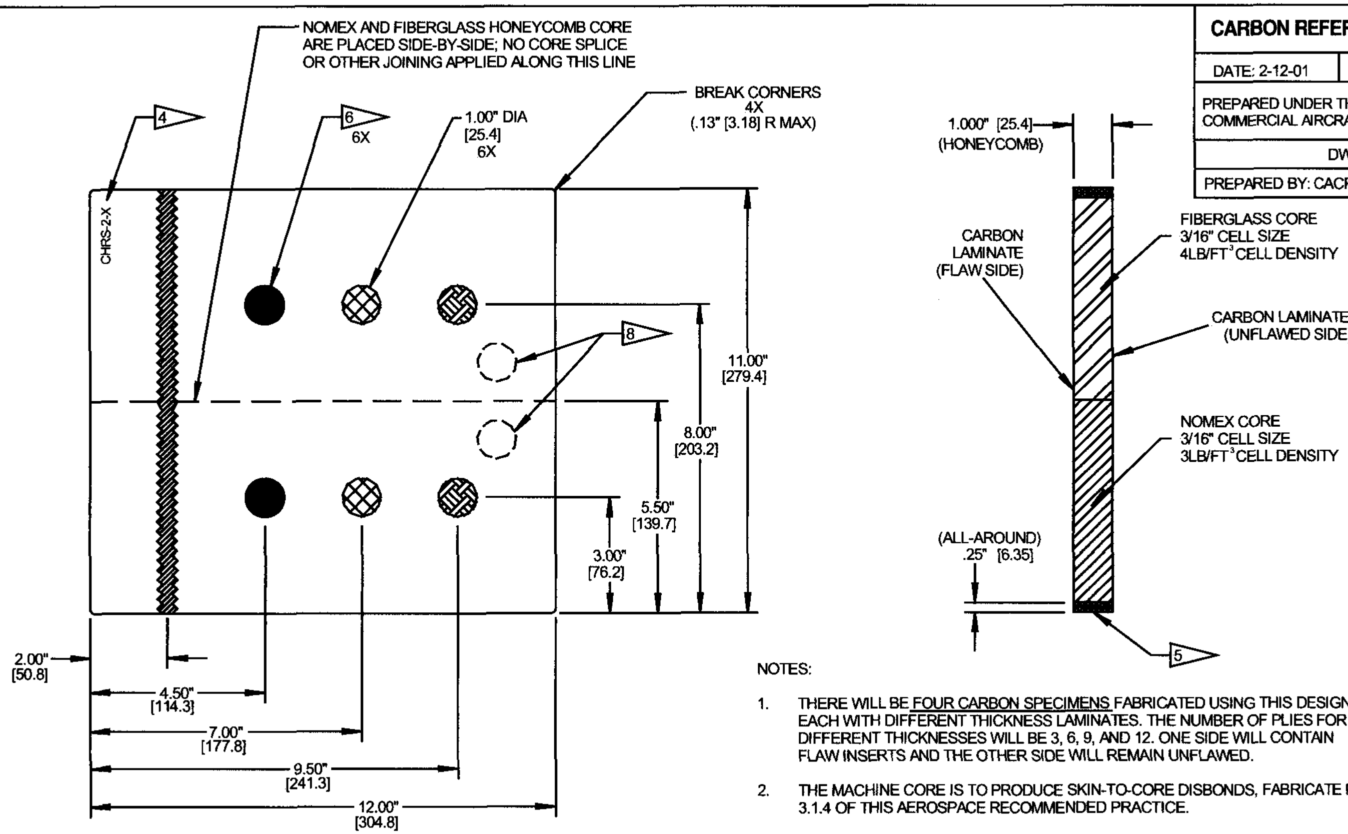
6. MARK THE TOP SURFACE OF SPECIMENS WITH 1.000" [25.4] DIA CIRCLE AT THE FLAW LOCATIONS. THESE MARKINGS (6X) SHOULD BE PERMANENT AND WITHIN ±.020 [.51] OF THE ACTUAL FLAW LOCATION.

7. DIMENSIONS ARE IN INCHES AND [ ] MILLIMETERS.

8. APPLY TWO PIECES OF CLOSED CELL ADHESIVE FOAM TAPE 1.00" [25.4] IN DIAMETER APPROXIMATELY AS SHOWN ON FARSIDE. THE TAPE WILL BE REMOVED AFTER THE SPECIMEN ACCEPTANCE TESTING (THROUGH-TRANSMISSION ULTRASONICS) IS COMPLETED. FOR MORE IMFORMATION SEE SECTION 3.2 IN THIS AEROSPACE RECOMMENDED PRACTICE.

9. TOLERANCES TO BE AS FOLLOWS UNLESS OTHERWISE SPECIFIED .XXX ±.010 [.254] AND .XX ±.05 [1.27] .

CARBON REFERENCE STANDARD		
DATE: 2-12-01	REV. #1	SCALE:1/2
PREPARED UNDER THE JURISDICTION OF ATA/IATA/SAE COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE		
DWG. NO. CHRS-2		
PREPARED BY: CACRC INSPECTION TASK GROUP		



MACHINE CORE (DEPTH OF 0.25" [6.35] IN HONEYCOMB)

PILLOW INSERTS (LOCATED TWO PLIES UP FROM CORE MATERIAL)

POTTED CORE

CORE SPLICE

NOTES:

1. THERE WILL BE FOUR CARBON SPECIMENS FABRICATED USING THIS DESIGN, EACH WITH DIFFERENT THICKNESS LAMINATES. THE NUMBER OF PLIES FOR THE DIFFERENT THICKNESSES WILL BE 3, 6, 9, AND 12. ONE SIDE WILL CONTAIN FLAW INSERTS AND THE OTHER SIDE WILL REMAIN UNFLAWED.

2. THE MACHINE CORE IS TO PRODUCE SKIN-TO-CORE DISBONDS, FABRICATE PER SECTION 3.1.4 OF THIS AEROSPACE RECOMMENDED PRACTICE.

3. PILLOW INSERTS ARE TO PRODUCE INTERPLY DELAMINATIONS, FABRICATE PER SECTION 3.1.4 OF THIS AEROSPACE RECOMMENDED PRACTICE.

4. MARK THE CARBON SKIN SPECIMEN NOS. USING A METHOD THAT IS PERMANENT. THE NOS. SHALL BE CHRS-2-3 FOR 3-PLY, CHRS-2-6 FOR 6-PLY, CHRS-2-9 FOR 9-PLY AND CHRS-2-12 FOR 12 PLY. LOCATE APPROXIMATELY AS SHOWN USING .25" [6.35] HIGH CHARACTERS.

5. NOTE AFTER CURING, THE EDGES OF HONEYCOMB ARE ROUTED TO MAKE DIMENSIONS LESS THAN THE LAMINATE (SKIN). THE EDGES ARE SEALED, AS PER SECTION 3.1.5 IN THIS AEROSPACE RECOMMENDED PRACTICE.

6. MARK THE TOP SURFACE OF SPECIMENS WITH A 1.000" [25.4] DIA CIRCLE AT THE FLAW LOCATIONS. THESE MARKING (6X) SHOULD BE PERMANENT AND WITHIN ±.020 [.51] OF THE ACTUAL FLAW LOCATION.

7. DIMENSIONS ARE IN INCHES AND IN [ ] MILLIMETERS.

8. APPLY TWO PIECES OF CLOSED CELL ADHESIVE FOAM TAPE 1.00" [25.4] IN DIAMETER APPROXIMATELY AS SHOWN ON FARSIDE. THE TAPE WILL BE REMOVED AFTER THE SPECIMEN ACCEPTANCE TESTING (THROUGH-TRANSMISSION ULTRASONICS) IS COMPLETED. FOR MORE INFORMATION SEE SECTION 3.2 IN THIS AEROSPACE RECOMMENDED PRACTICE.

9. TOLERANCES TO BE AS FOLLOWS UNLESS OTHERWISE SPECIFIED .XXX ±.010 [.254] AND .XX ±.05 [.127] .

3.4 **Forms**

As-Built Form For Composite Honeycomb Reference Standards

PANEL NUMBER/DESCRIPTION ________________________________

GENERAL

Honeycomb Panels Labeled by Specimen Number on Side with Flaws? ________
Smooth (tool side) of Laminate Facing out on Honeycomb Sandwich? _________
Honeycomb Oriented with Core Ribbon along 0° Axis? _____
Honeycomb Oriented with Machined Core Facing Down During Laminate-to-Core Bonding? ________

MATERIALS USED

Laminate Material Type: ______________________________________
Number of Plies: ____________
Honeycomb Material Type: __________________________________
Weight of Honeycomb: ____________
Adhesive for Laminate-to-Honeycomb Bond Cure:________________________
Potted Core Material: ____________
Core Splice Material: ____________
Edge Sealant Material:____________

LAMINATE CURE TEMPERATURE PROFILE

Ref. Aviation Industry Specification: ________________________ (if applicable)

Target Cure Temperature (degrees F/C):_______________
Target Dwell Time (minutes):_______________
Allowable Heat-up Rate:_____________
Allowable Cool-down Rate:_______________

Beginning Temperature (degrees F/C): ___________
Beginning Time (minutes): ___________

Elapsed Time to Target Cure Temperature (minutes): ___________
Maximum Temperature During Dwell Time (degrees F/C): _____________
Minimum Temperature During Dwell Time (degrees F/C): _____________
Elapsed Time at Dwell Temperature (minutes): ___________
Elapsed Time from End of Dwell to End of Cool-down (minutes): _____________

Ending Temperature(F/C): ___________
Ending Time (minutes): _____________

LAMINATE-TO-HONEYCOMB CURE TEMPERATURE PROFILE

Ref. Aviation Industry Specification: ____________________ (if applicable)

Target Cure Temperature (degrees F/C):____________
Target Dwell Time (minutes):____________
Allowable Heat-up Rate:____________
Allowable Cool-down Rate:____________

Beginning Temperature(degrees F/C): ____________
Beginning Time (minutes): ____________

Elapsed Time to Target Cure Temperature (minutes): ____________
Maximum Temperature During Dwell Time (degrees F/C): ____________
Minimum Temperature During Dwell Time (degrees F/C): ____________
Elapsed Time at Dwell Temperature (minutes): ____________
Elapsed Time from End of Dwell to End of Cool-down (minutes): ____________

Ending Temperature (degrees F/C): ____________
Ending Time (minutes): ____________

CURE PRESSURE

Bag Pressure During Laminate Cure:____________
Bag Pressure During Laminate-to-Honeycomb Bond Cure:____________

NOTES

____
____
____
____
____
____

Fabrication Performed By:	**Date:**

4.0 NOTES

4.1 Materials

The following industry specifications and material designations listed in this Recommended Practice are for information purposes. Such listings shall not be construed as an endorsement or guarantee of performance by SAE.

A. Plain Weave, Pre-Preg, 350° Carbon Graphite Cloth:
- BMS 8-256 Plain Weave Graphite
  - → Industry Descriptor: Class 2, Type IV, 3K-70-PW:
    - HMF 970/PWC(TY) (Cytec Fiberite, U.S.A.)
    - W3T-282(Y)-XX-F593-18 (Hexcel, U.S.A. & Japan)
- A.0086/00
- 6814NT
- ABR1-0009, 1-0013, 1-0026

B. Plain Weave, Pre-Preg, 250° Fiberglass Cloth:
- BMS 8-79 1581
- BMS 8-79 7781
- 1581-F155-5-F69 (Hexcel, U.S.A. & Japan)
- 1581-F155-5-CS272 (Hexcel, Belgium)
- MXB7701/1581-Z6040 (Kasei Composites, Japan)
- MXB7701-1581-B3 (Cytec Fiberite, U.S.A.)
- G1581/F6986S03-S920NM (Yokohama Rubber, Japan)
- HG120/RS1212-Z6040 (Han Kuk Fiberglass, Korea)

C. Honeycomb:1 inch thick.
- Fiberglass 3/16 inch cell size, 4 lb/ft^3 density
- Nomex 3/16 inch cell size, 3lb/ft^3 density

D. Adhesive: 2 plies (each side) of 0.005" (0.13mm) adhesive or 1 ply of 0.010" (.26mm) adhesive; type for 225°F to 250°F (107°C to120°C) secondary bond.
- BMS 5-101 Grade 10
- AF163 epoxy film

E. Potting Material (Potted Core):
- CG-1305 epoxy (Ciba-Geigy)
- BMS 5-28 Type 7
- BM 3500 B/A
- ABR 2-0055
- Cytec Fiberite BR 623PR-5LTR
- Scotch Weld EC3439 HAT-AF
- Stycast 1090SI

F. Foaming Adhesive (Core Splice):
- BMS 5-90
- AF-3028
- FM410-1.050
- AF 3024.050
- L 657.050
- ABR 2-0049

G. Sealant:
- BMS 5-28 Type 7 with 1305 Resin/Hardener; recommend adding chopped fiberglass to make resin more viscous and easier to apply around entire perimeter at one time.
- Stycast 1090SI
- Ciba Araldite 2020 A/B
- EA 9395
- Any epoxy type sealant that will produce a watertight seal around the perimeter. Transparent seals are preferred as they allow honeycomb type to be visible.

H. Tissue Paper:
- Thin paper such as used for tracing: 0.002 inches (.05mm) thick.

I. Heat Resistant, Polymide Film Tape:
- 0.002 inches (.05mm) thick (e.g. Kapton tape)

J. Foam Tape:
- Closed Cell Vinyl Foam Tape (e.g. 3M part no. 4416)

4.2 **Keywords**: Non-destructive inspection, composite honeycomb, reference standards, NDI.

5.0 REFERENCES

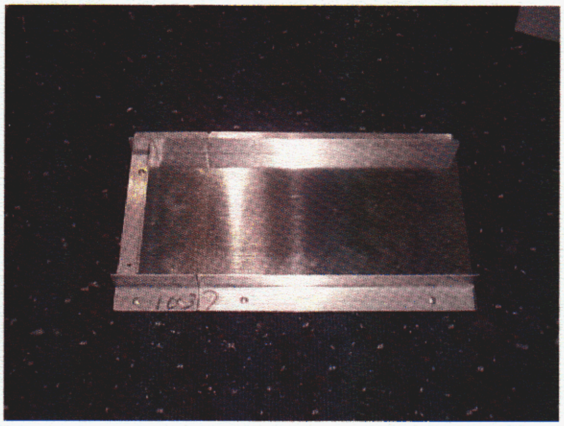
1. Roach, D.P., Dorrell, L.R., Kollgaard, J., Dreher, T., "Improving Aircraft Composite Inspections Using Optimized Reference Standards", SAE Airframe Maintenance and Repair Conference, Nov. 1998, SAE Technical Paper 98AEMR-34

PREPARED UNDER THE JURISDICTION OF ATA/IATA/SAE
COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE (CACRC)

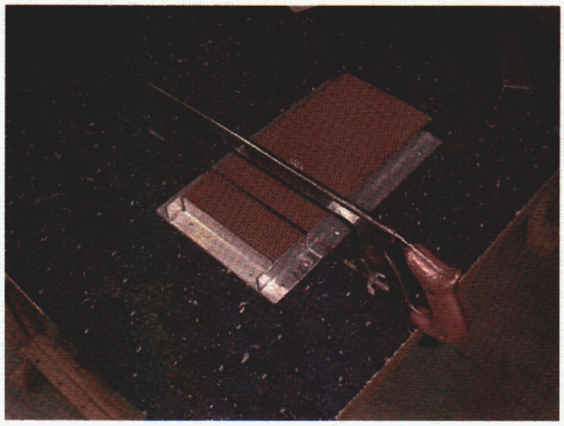
Appendix A of ARP 5606

Photographs Showing Honeycomb Reference Standard Panel Production Activities

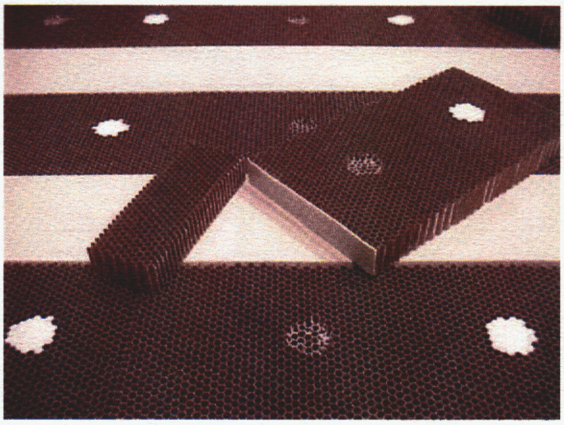
HONEYCOMB CORE PREPARATION



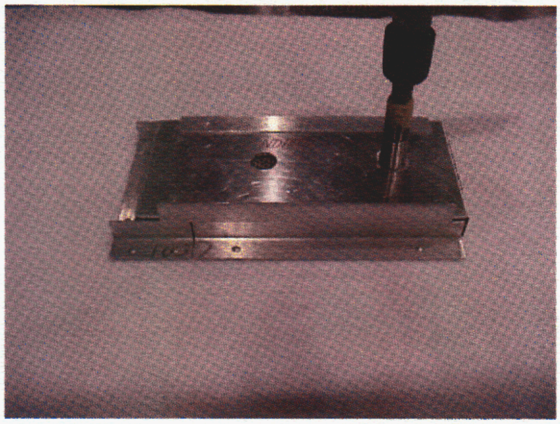
1. Fixture for cutting core splice specimen



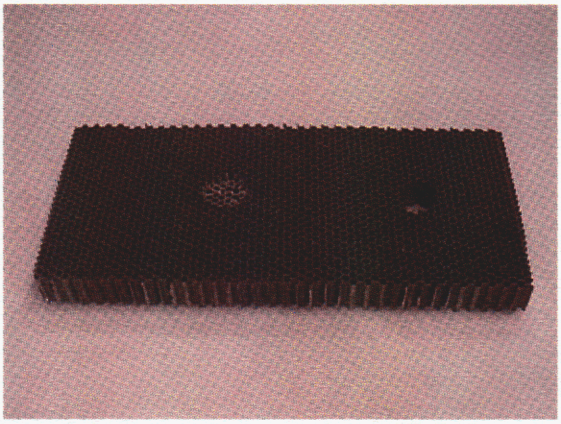
2. Cutting core material for core splice



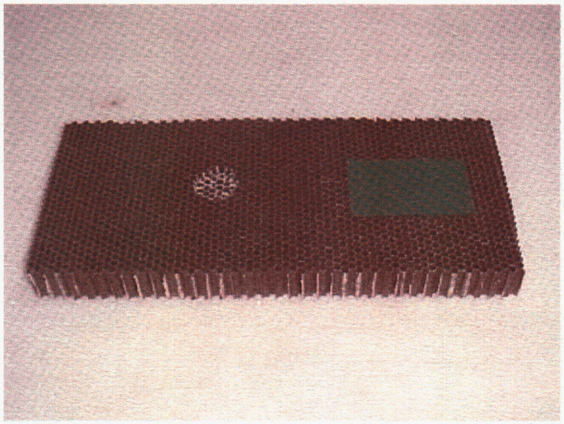
3. Core splice - single strip of foaming adhesive



4. Fixture for removing 1" diameter of core for potted core plug; can also fill individual cells instead of removing core

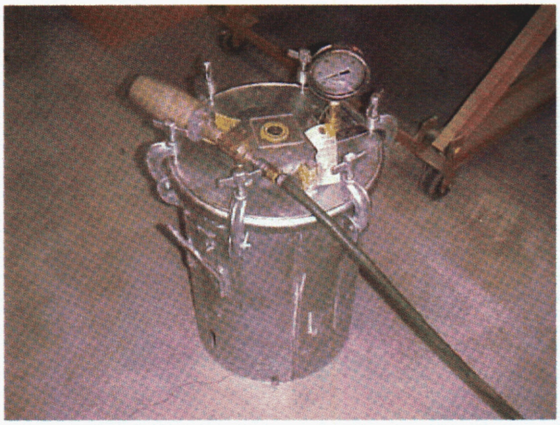


5. One inch diameter hole for potted core

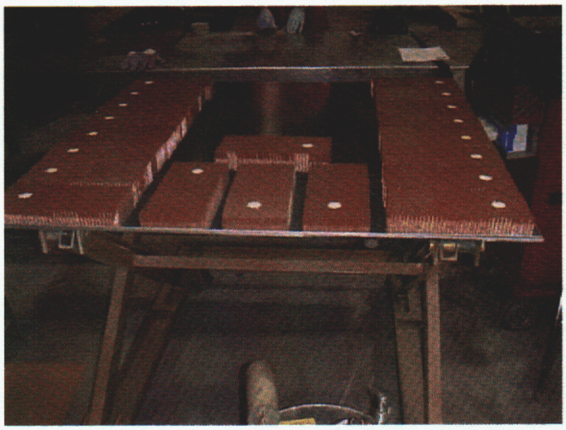


6. Tape on back side of potted core area to retain potting material

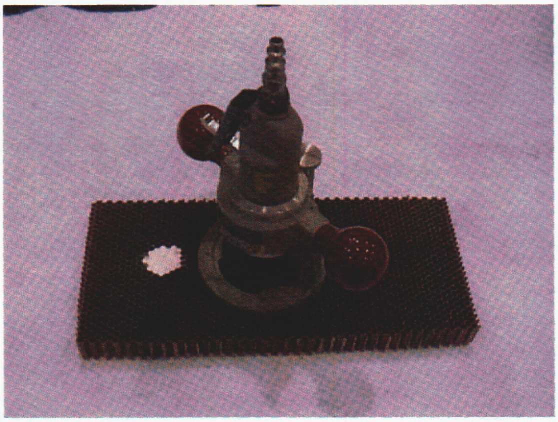
HONEYCOMB CORE PREPARATION (continued)



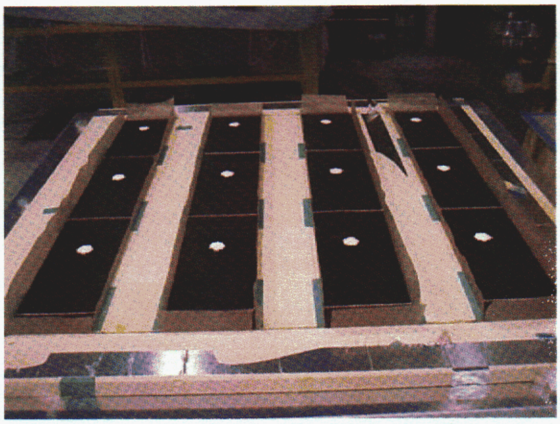
7. Vacuum pot for core potting compound



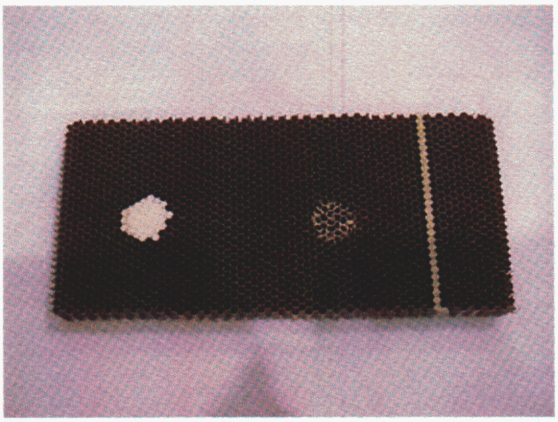
8. Nomex honeycomb filled with core potting material and place on table, flawed side down, for curing



9. Potted area machined smooth with diamond bit router

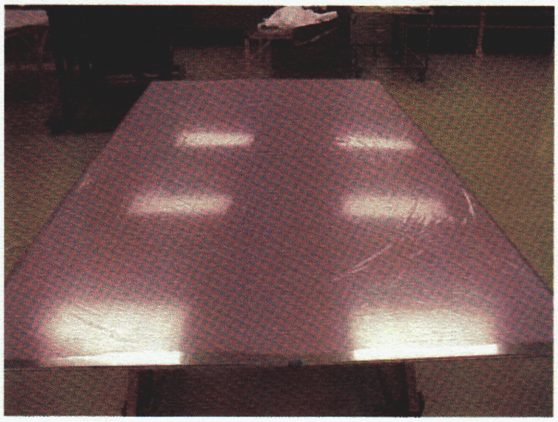


10. Fiberglass honeycomb after core splice and potted core materials have been cured at 250°F for 120 minutes

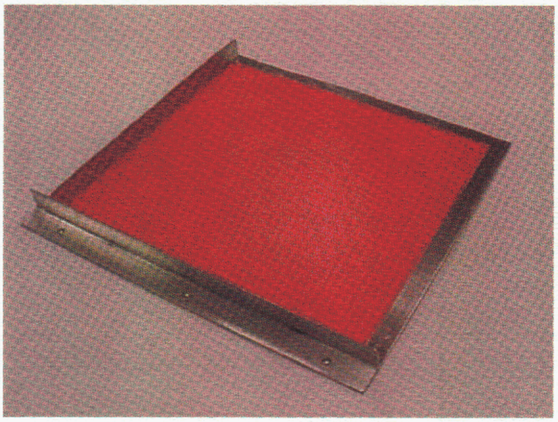


11. Completed honeycomb core showing potted core, machined core, and core splice areas

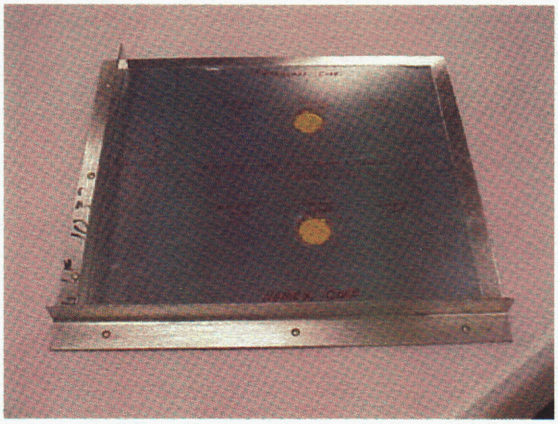
LAMINATE SKIN PREPARATION



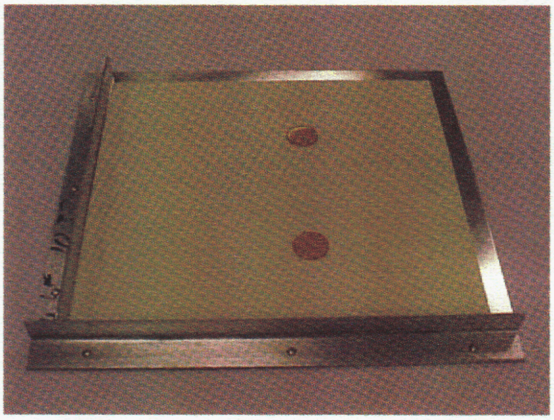
1. Solid release film on bonding tool



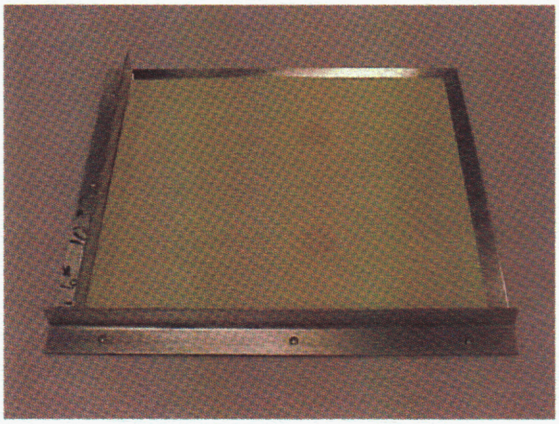
2. Fixture for assembling laminates



3. Template for positioning flaws in the laminates

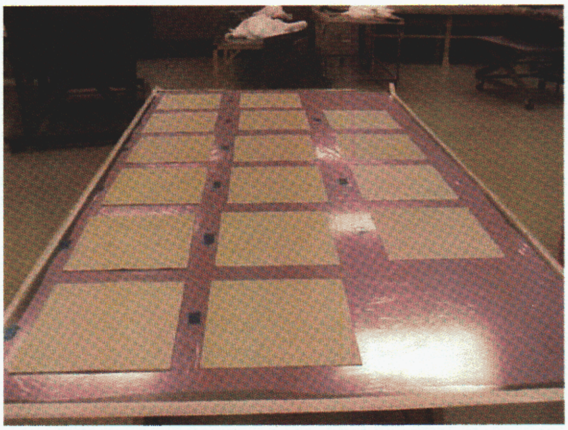


4. Fiberglass plies being laid up with Pillow Insert delamination flaws

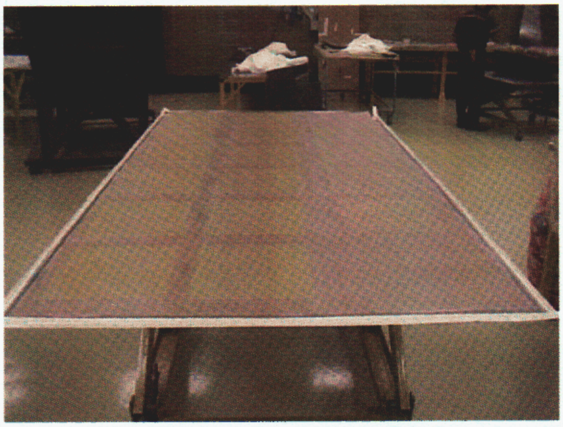


5. Uncured laminate ready for vacuum bag

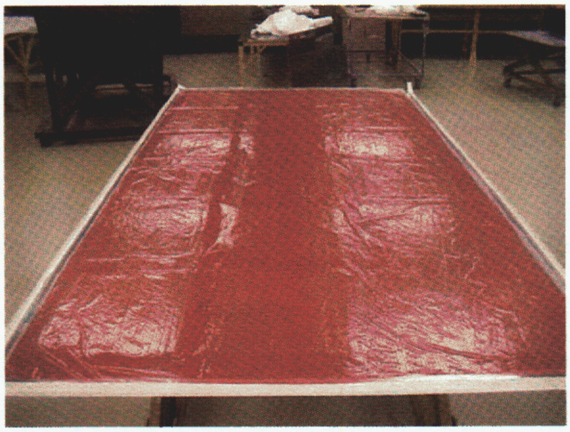
LAMINATE SKIN PREPARATION (continued)



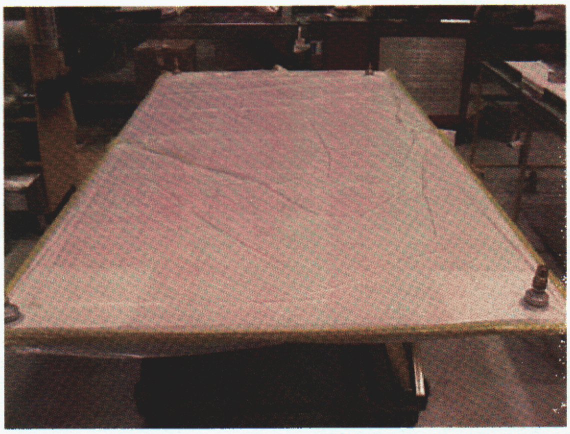
6. Uncured laminates on bonding table



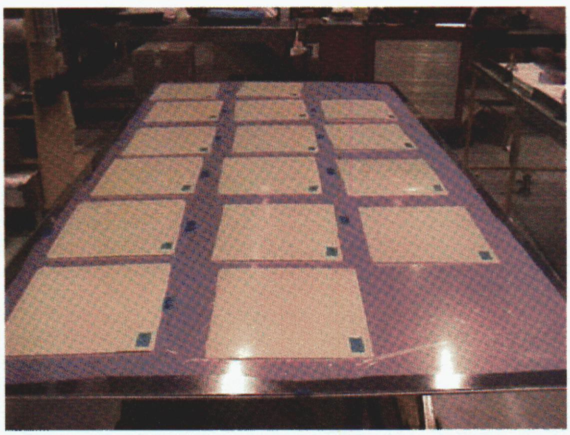
7. Teflon release film placed over laminates



8. Perforated release film covers the Teflon sheet

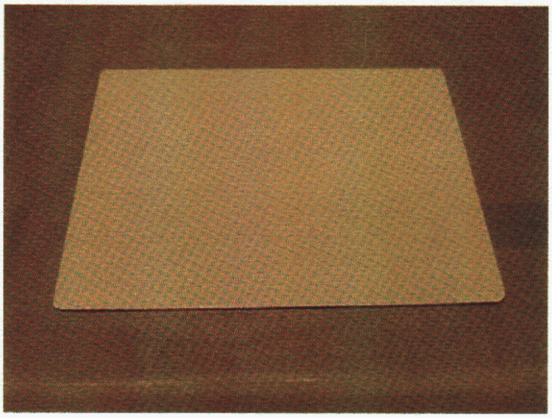


9. Bleeder cloth and bagging material; 11- 12 psi (75.9 - 82.7 kPa) vacuum applied to all laminates

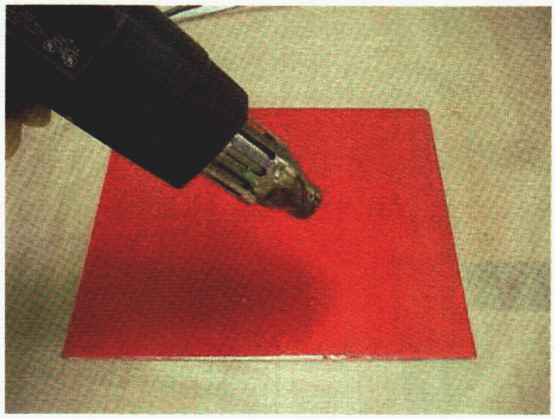


10. Cured laminate skins for 8 reference standards;8 front (with flaws) and 8 back (unflawed) skins

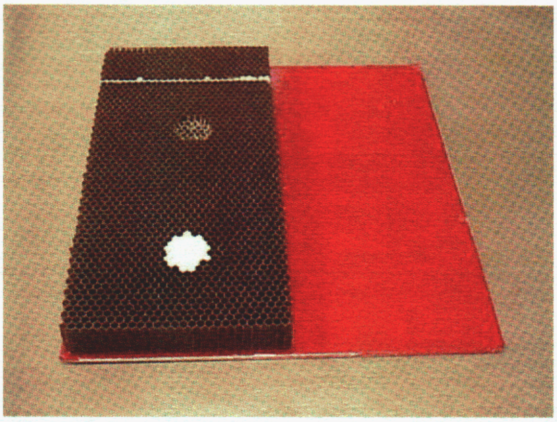
HONEYCOMB PANEL ASSEMBLY



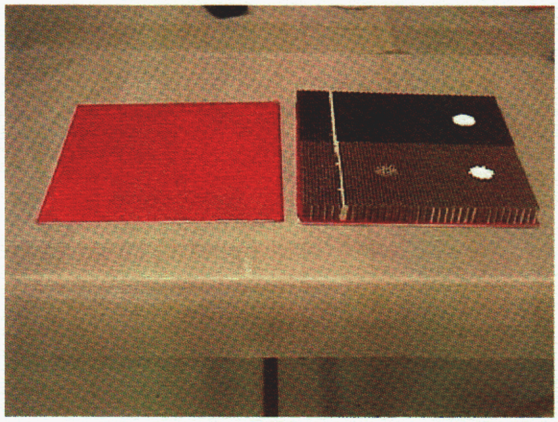
1. Cured laminate skin



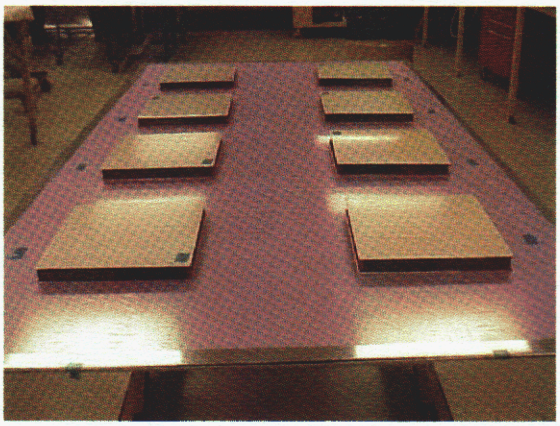
2. Adhesive sheet added to laminate



3. Nomex core is located on laminate skin; fiberglass core to follow



4. Adhesive layer placed on flawed laminate with both cores in place on lower laminate skin



5. Specimens placed on bonding table, flawed side down, for vacuum bag assembly and cure process

APPENDIX B

Aerospace Recommended Practice 5605: Solid Composite Laminate NDI Reference Standards

Aerospace Recommended Practice

Solid Composite Laminate NDI Reference Standards

TABLE OF CONTENTS

Nomenclature:

ARP	Aerospace Recommended Practice
NDI	Nondestructive Inspection
NDT	Nondestructive Testing
NEMA	National Electric Manufacturers Association
OEM	Original Equipment Manufacturer

1. SCOPE

This recommended practice establishes a generic composite reference standard that will accommodate nondestructive inspections (NDI) on the full array of glass fiber and carbon fiber laminates found on aircraft.

1.1 Purpose

The purpose of this Aerospace Recommended Practice (ARP) is to describe the design and production of solid composite laminate calibration standards to be used in ultrasonic, resonant, and tap test NDI equipment calibration for accomplishment of damage assessment and post-repair inspections. It is intended that these standards be adopted by aircraft Original Equipment Manufacturers within procedures contained in their Nondestructive Testing Manuals. Depending on the nature of the inspection, it may be necessary to compensate for variations in material properties through the use of correction factors or by adjusting for these differences on the part or structure being inspected. When using these standards consideration must be given to surface coatings such as paint or lightning protection plies. This is a reference standard construction document and not an inspection document. Inspection procedures, from OEM or users' maintenance manuals, must accompany the use of these reference standards for each unique family of composite laminate construction.

1.2 Background

The CACRC Inspection Task Group developed this ARP in an effort to establish a single, generic set of composite laminate reference standards that would accommodate inspections on the full array of laminates found on aircraft. The advantages of industry-wide acceptance of these composite standards include: 1) provides a consistent approach to composite inspections thus improving inspection reliability, 2) reduces standard procurement costs, and 3) aids in future assessments of composite inspection technologies. Specific use of the laminate standards described in this ARP can be achieved through the OEM inspection procedures found in Nondestructive Testing Manuals and Nondestructive Testing Standard Practice Manuals.

1.3 Supporting Data

Through-transmission ultrasonics was applied to the series of existing Boeing, Douglas, and Airbus laminate standards, as well as, material samples gathered from a variety of sources, in order to measure the velocity and relative attenuation properties in the laminates. Acoustic impedance for these materials was then derived using measured velocity and material density values. Upper and lower bounds for these key material properties were established for both carbon graphite (tape and fabric) and fiberglass laminates. A material search identified G11 Phenolic as a material that has velocity, acoustic impedance, and attenuation values that closely matched those of carbon graphite and fiberglass. More precisely, the G11 properties are midway between graphite and fiberglass laminates. While G11 is not an exact match with either carbon graphite or fiberglass materials, it was decided that G11 is ideally suited for use as a generic standard because of the normal variations ($\pm$ 10%) found within laminate specimens, as well as, variations reported during field inspections on similar laminate structures. In addition, it was found that the consistency of G11 Phenolic material from one batch to the next was much better than the consistency observed in either carbon graphite or fiberglass standards previously fabricated. Finally, G11 Phenolic standards can improve on existing solid laminate standards since the material is inexpensive, can be reliably manufactured and is easy to machine into a solid laminate standard (i.e. plate with multiple thicknesses). Prototype laminate standards were fabricated from the G11 material and inspection data was accumulated to validate the use of G11 standards for use in carbon and fiberglass laminate inspections.

2. APPLICABLE DOCUMENTS

The following publications form a part of this specification to the extent specified herein. The applicable issue of the referenced publications shall be the issue in effect on the date of the purchase order.

2.1 U. S. Government Publications

Available from DODSSP, Subscription Services Desk, Building 4D, 700 Robbins Avenue, Philadelphia, PA 19111-5094.

MIL-L 24768 Insulation, Plastics, Laminated, Thermosetting, General Specification for

MIL-I-24768/3 Insulation, Plastic, Laminated, Thermosetting, Glass Cloth, Epoxy Resin (GEB)

2.2 NEMA Publications

Available from NEMA, 1300 North 17th Street, Suite 1847, Rosslyn, Virginia 22209

LI 1-1998 Industrial Laminated Thermosetting Products (Grade G11)

3.0 TECHNICAL REQUIREMENTS

3.1 Fabrication and Material

Fabrication of the solid composite laminate standards (See attached engineering design drawings G11-STD-1, G11-STD-2, G11-STD-3, and G11-STD-4) is straightforward and consists of three machining tasks: 1) face the upper and lower surfaces of a thick sheet of G11 Phenolic material [1/] such that the two surfaces are flat and parallel and the resulting plate thickness will allow for the steps shown in the drawings, 2) cut the Phenolic plate into four 5.75 inches [146 mm] H X 4 inches [101.6 mm] W bricks, and 3) machine flat-bottomed holes into the bricks to produce the thickness steps shown in the engineering drawings. The machined plates should be lapped to make a smooth inspection surface that mimics those found on aircraft laminate structure. Finally, the thickness designations should be labeled and the flat-bottomed holes should be marked with a circle to aid inspection probe placement.

Figure 1 shows schematics of the solid laminate standards while Figure 2 contains a photo of the prototype laminate standard set. Key issues addressed by the designs are as follows: 1) protection against moisture ingress - extensive exposure to water submersion showed that water absorption is not a problem with G11 material, 2) locating probe - the location of each skin thickness has been identified on the laminates to allow for proper positioning of the transducer and each thickness has been labeled, 3) surface finish - the surface finish is improved via a lapping process to produce more consistent responses from transducers, 4) size and ease of handling - the set of 24 thicknesses has been distributed over four different plates, and 5) one ply resolution - the thinnest skin was reduced from 0.010 inch thick to 0.007 inch thick to closer represent one ply.

The starting thickness of Phenolic plate should be at least 1 inch [25.4 mm] thick for the thickest laminate standard (G11-STD-4) with thickness steps ranging from 0.5 inch [12.7 mm] to 1.0 inch [25.4 mm]. For the other three laminate standards (G11-STD-1 through G11-STD-3) with thickness steps ranging from 0.007 inch [0.18 mm] to 0.45 inch [11.43 mm], the starting thickness of the G11 Phenolic plate should be 0.5 inch [12.7 mm]. This will produce the most uniform standards with responses that closely match fiberglass and carbon laminates.

1/ Use NEMA Grade G11 fabricated in accordance with MIL-I-24768/3 Type GEB. Do not use NEMA Grade G11/FR5 (fire retardant) which is fabricated in accordance with MIL-I-24768/28 Type GEB-F

3.2 Acceptance Criteria

- The G11 Laminate Reference Standards must be certified by a series of mechanical thickness measurements. The material thickness in each flat-bottomed hole must be measured using micrometer or other thickness measurement device that is traceable to primary or secondary standards and has a resolution of 0.001 inch [0.0254 mm]. Measurements shall be made at a minimum of three places within the hole, spaced approximately 1/2" [12.7 mm] apart, and recorded. All measurements shall meet the required thickness tolerance callout.

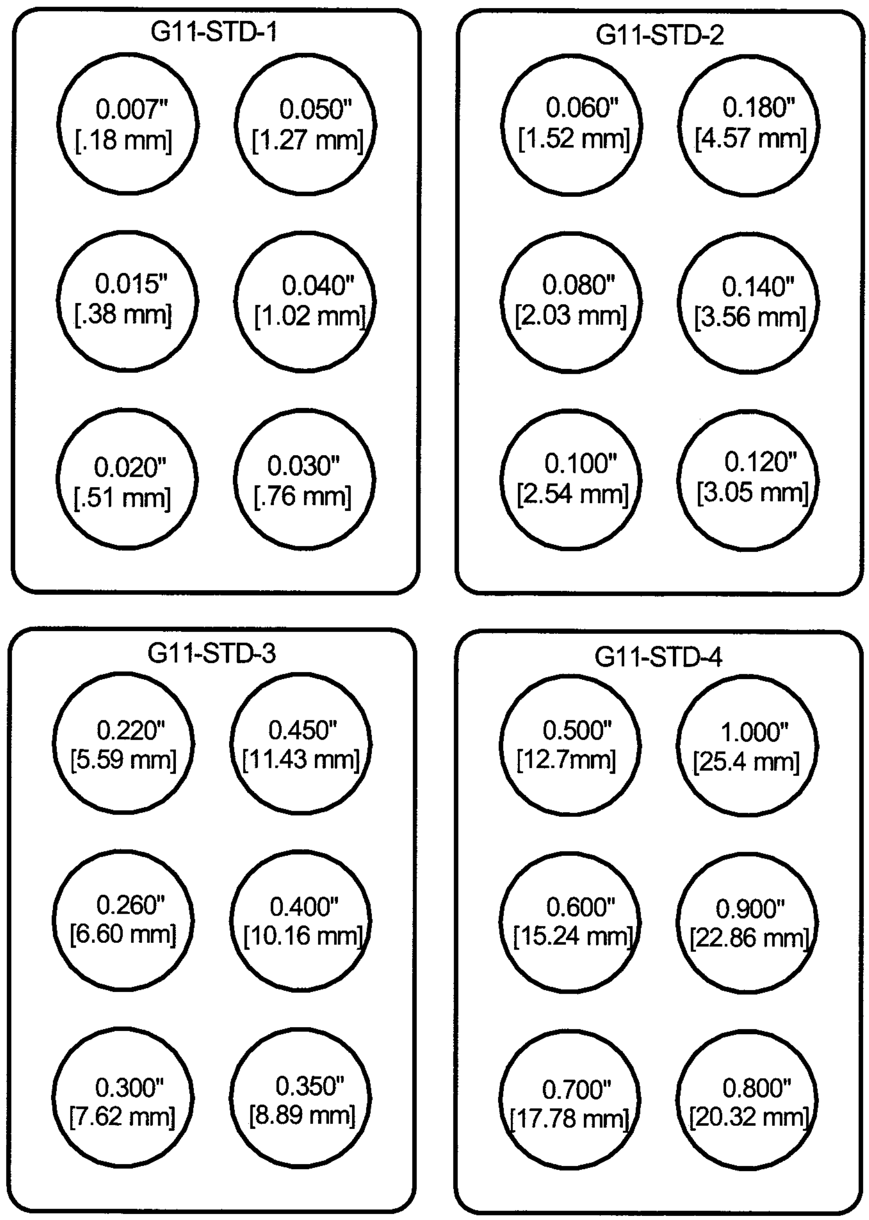


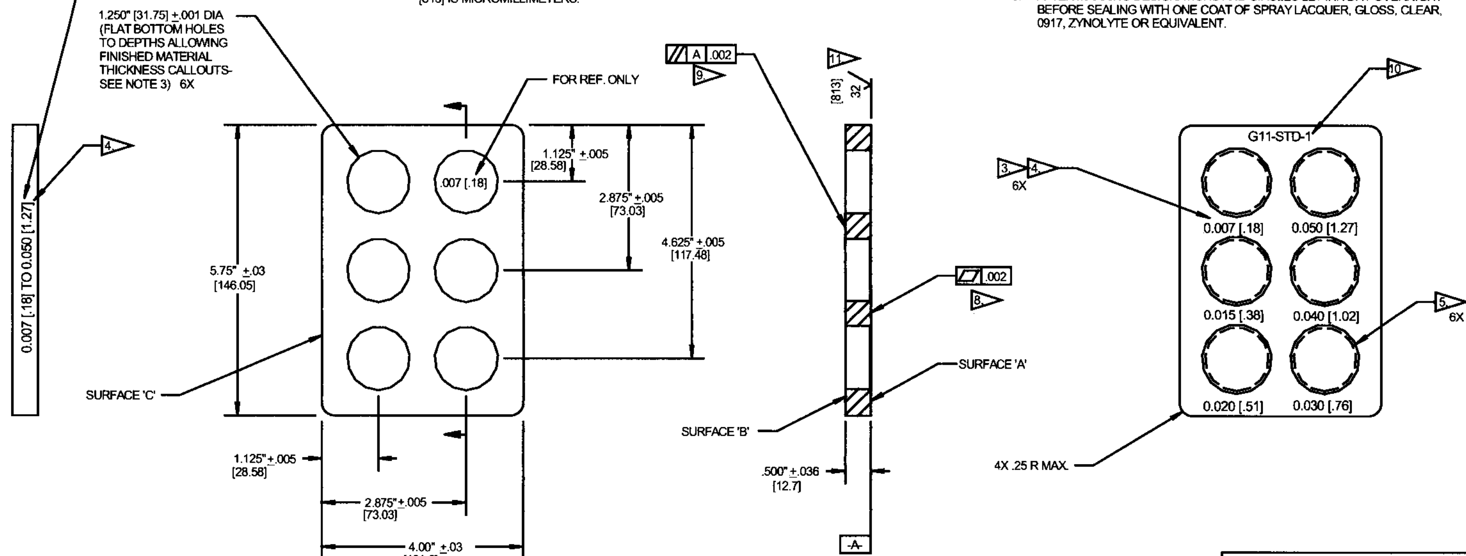
Figure 1: G11 Solid Laminate Standard Set
(numbers in circles represent skin thickness, in inches [or mm in brackets], of each flat bottom hole)

Figure 2: Front and Back Photos of G11 Solid Laminate Set

3.3 Engineering Design Drawings for Composite Laminate Reference Standards are as shown in G11-STD-1, G11-STD-2, G11-STD-3, and G11-STD-4.

NOTES:

1. MATERIAL: G11 PHENOLIC FABRICATED IN ACCORDANCE WITH Mil-I-24768/3, TYPE GEB, .500" [12.7] THK.
2. DIMENSIONS ARE IN INCHES AND NUMBERS IN [ ] ARE MILLIMETERS.
3. THE VALUES SHOWN REPRESENT THE THICKNESS OF MATERIAL LEFT IN THE FLAT BOTTOM HOLES AFTER THE MACHINING PROCESS WITHIN ±.001 [.025] .
4. MARK DESIGNATIONS WITH INK USING LEROY LETTERING GUIDE OR EQUIVALENT. LOCATE APPROXIMATELY AS SHOWN USING .200" [5.08] HIGH CHARACTERS.
5. MARK A 1.375" [34.93] DIA CIRCLE CENTERED AROUND EACH FLAT BOTTOM HOLE ON THE FLAT SIDE WITH INK USING TEMPLATE. (USE INK THAT WILL NOT SMEAR WHEN SPRAYED WITH FINISH COAT).
6. AFTER MARKING DESIGNATIONS AND CIRCLES LET INK DRY OVERNIGHT BEFORE SEALING WITH ONE COAT OF SPRAY LACQUER, GLOSS, CLEAR, 0917, ZYNOLYTE OR EQUIVALENT.
7. THE MATERIAL THICKNESS IN THE FLAT BOTTOM HOLES MUST BE INSPECTED AT A MINIMUM OF THREE PLACES WITHIN THE HOLE AND RECORDED. ALL MEASUREMENTS SHALL MEET THE REQUIRED TOLERANCE CALLOUT, IF THIS IS NOT THE CASE THE PART WILL BE REJECTED.
8. SURFACE 'A' SHALL BE FLAT WITHIN .002 [.051]. THIS FLATNESS SPECIFIES A TOLERANCE ZONE DEFINED BY TWO PARALLEL PLANES WITHIN WHICH THE SURFACE MUST LIE.
9. SURFACE 'B' TO BE PARALLEL TO SURFACE 'A' WITHIN .002 [.051]. THIS PARALLELISM IS A TOLERANCE ZONE DEFINED BY TWO PLANES PARALLEL TO SURFACE 'A' WITHIN WHICH THE LINE ELEMENTS OF THE SURFACE MUST LIE.
10. MARK PART NUMBER WITH INK USING LEROY LETTERING GUIDE OR EQUIVALENT. LOCATE APPROXIMATELY AS SHOWN USING .200" [5.08] HIGH CHARACTERS.
11. THE SURFACE FINISH NUMBER (32) REPRESENTS MICROINCHES WHILE THE [813] IS MICROMILLIMETERS.



TOLERANCE METRIC EQUIVALENTS	
INCHES	MILLIMETERS
.001	[.025]
.005	[.127]
.030	[.762]
.036	[.914]

G11 STANDARD - 0.050"

Drawing No. G11-STD-1

Scale: 1/2 Date: 10-10-00

PREPARED UNDER JURISDICTION OF ATA/IATA/SAE COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE

PREPARED BY: CACRC INSPECTION TASK GROUP

NOTES:

1. MATERIAL: G11 PHENOLIC FABRICATED IN ACCORDANCE WITH Mil-I-24768/3, TYPE GEB, .500" [12.7] THK.

2. DIMENSIONS ARE IN INCHES AND NUMBERS IN [ ] ARE MILLIMETERS.

3. THE VALUES SHOWN REPRESENT THE THICKNESS OF MATERIAL LEFT IN THE FLAT BOTTOM HOLES AFTER THE MACHINING PROCESS WITHIN ±.001 [.025].

4. MARK DESIGNATIONS WITH INK USING LEROY LETTERING GUIDE OR EQUIVALENT. LOCATE APPROXIMATELY AS SHOWN USING .200" [5.08] HIGH CHARACTERS.

5. MARK A 1.375" [34.93] DIA CIRCLE CENTERED AROUND EACH FLAT BOTTOM HOLE ON THE FLAT SIDE WITH INK USING TEMPLATE. (USE INK THAT WILL NOT SMEAR WHEN SPRAYED WITH FINISH COAT).

6. AFTER MARKING DESIGNATIONS AND CIRCLES LET INK DRY OVERNIGHT BEFORE SEALING WITH ONE COAT OF SPRAY LACQUER, GLOSS, CLEAR, 0917, ZYNOLYTE OR EQUIVALENT.

7. THE MATERIAL THICKNESS IN THE FLAT BOTTOM HOLES MUST BE INSPECTED AT A MINIMUM OF THREE PLACES WITHIN THE HOLE AND RECORDED. ALL MEASUREMENTS SHALL MEET THE REQUIRED TOLERANCE CALLOUT, IF THIS IS NOT THE CASE THE PART WILL BE REJECTED.

8. SURFACE 'A' SHALL BE FLAT WITHIN .002 [.051]. THIS FLATNESS SPECIFIES A TOLERANCE ZONE DEFINED BY TWO PARALLEL PLANES WITHIN WHICH THE SURFACE MUST LIE.

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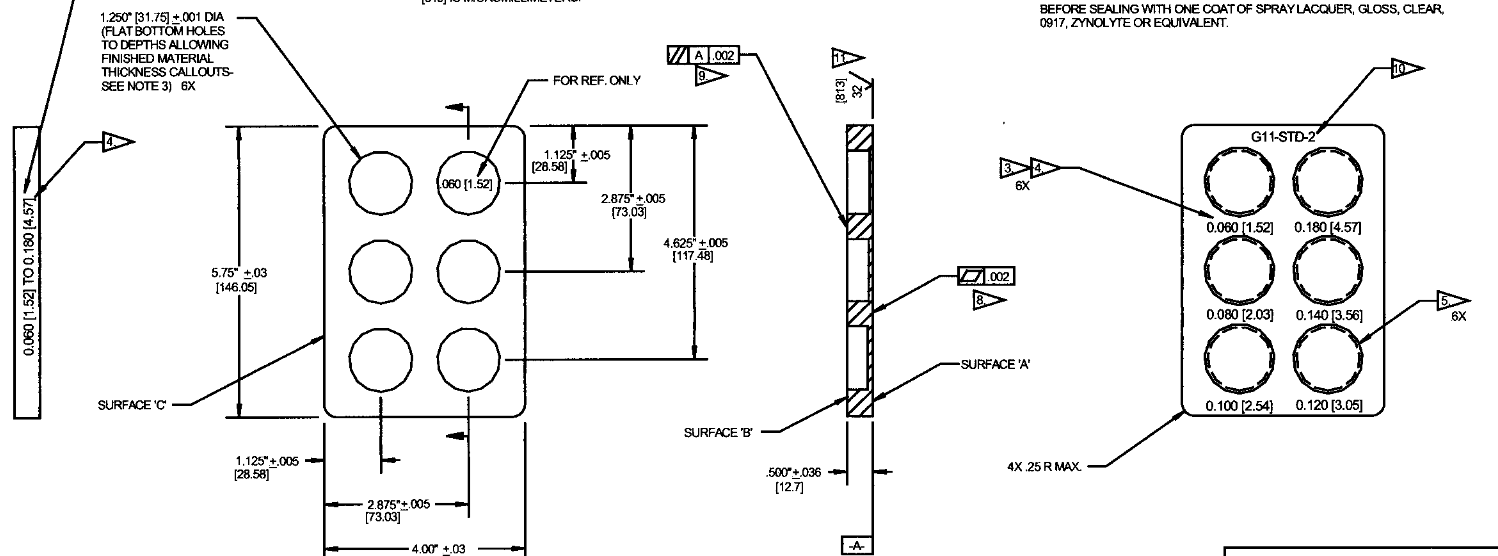

TOLERANCE METRIC EQUIVALENTS	
INCHES	MILLIMETERS
.001	[.025]
.005	[.127]
.030	[.762]
.036	[.914]

G11 STANDARD - 0.180"

Drawing No. G11-STD-2

Scale: 1/2 Date: 10-10-00

PREPARED UNDER JURISDICTION OF ATA/IATA/SAE COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE

PREPARED BY: CACRC INSPECTION TASK GROUP

NOTES:

1. MATERIAL: G11 PHENOLIC FABRICATED IN ACCORDANCE WITH Mil-I-24768/3, TYPE GEB, .500" [12.7] THK.
2. DIMENSIONS ARE IN INCHES AND NUMBERS IN [ ] ARE MILLIMETERS.
3. THE VALUES SHOWN REPRESENT THE THICKNESS OF MATERIAL LEFT IN THE FLAT BOTTOM HOLES AFTER THE MACHINING PROCESS WITHIN ±.001 [.025].
4. MARK DESIGNATIONS WITH INK USING LEROY LETTERING GUIDE OR EQUIVALENT. LOCATE APPROXIMATELY AS SHOWN USING .200" [5.08] HIGH CHARACTERS.
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6. AFTER MARKING DESIGNATIONS AND CIRCLES LET INK DRY OVERNIGHT *BEFORE SEALING WITH ONE COAT OF SPRAY LACQUER, GLOSS, CLEAR,* 0917, ZYNOLYTE OR EQUIVALENT.
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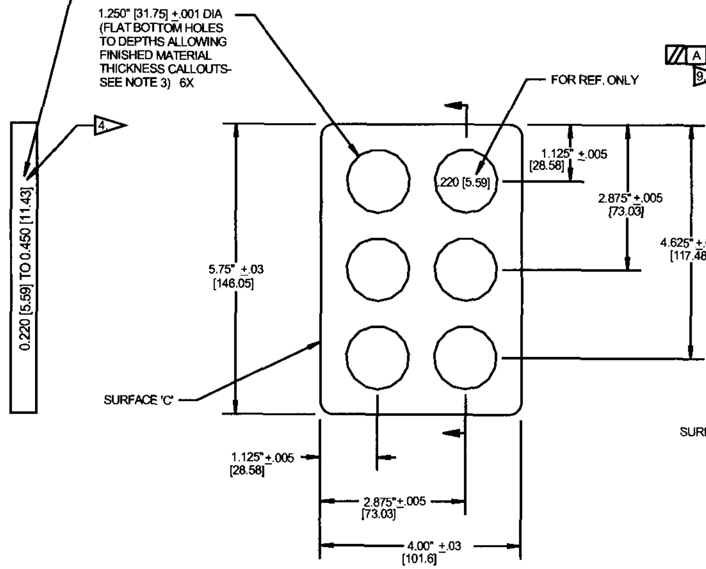
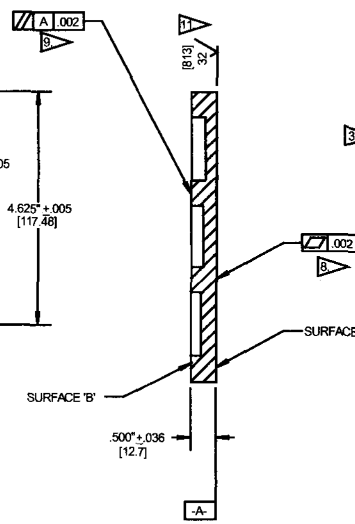
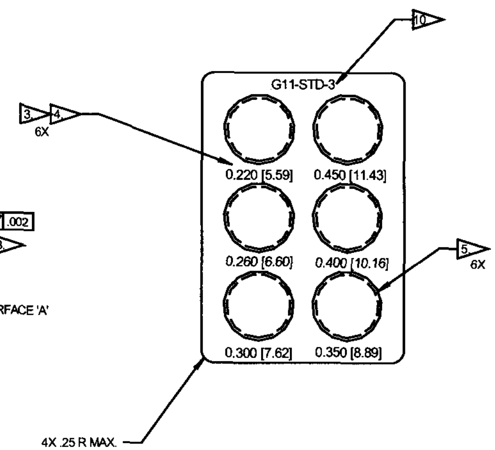






TOLERANCE METRIC EQUIVALENTS	
INCHES	MILLIMETERS
.001	[.025]
.005	[.127]
.030	[.762]
.036	[.914]

G11 STANDARD - 0.450"

Drawing No. G11-STD-3

Scale: 1/2 Date: 10-10-00

PREPARED UNDER JURISDICTION OF ATA/IATA/SAE COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE

PREPARED BY: CACRC INSPECTION TASK GROUP

12. THE SURFACE FINISH NUMBER (32) REPRESENTS MICROINCHES WHILE THE [813] IS MICROMILLIMETERS.

7. THE MATERIAL THICKNESS IN THE FLAT BOTTOM HOLES MUST BE INSPECTED AT A MINIMUM OF THREE PLACES WITHIN THE HOLE AND RECORDED. ALL MEASUREMENTS SHALL MEET THE REQUIRED TOLERANCE CALLOUT, IF THIS IS NOT THE CASE THE PART WILL BE REJECTED.

8. SURFACE 'A' SHALL BE FLAT WITHIN .002 [.051]. THIS FLATNESS SPECIFIES A TOLERANCE ZONE DEFINED BY TWO PARALLEL PLANES WITHIN WHICH THE SURFACE MUST LIE.

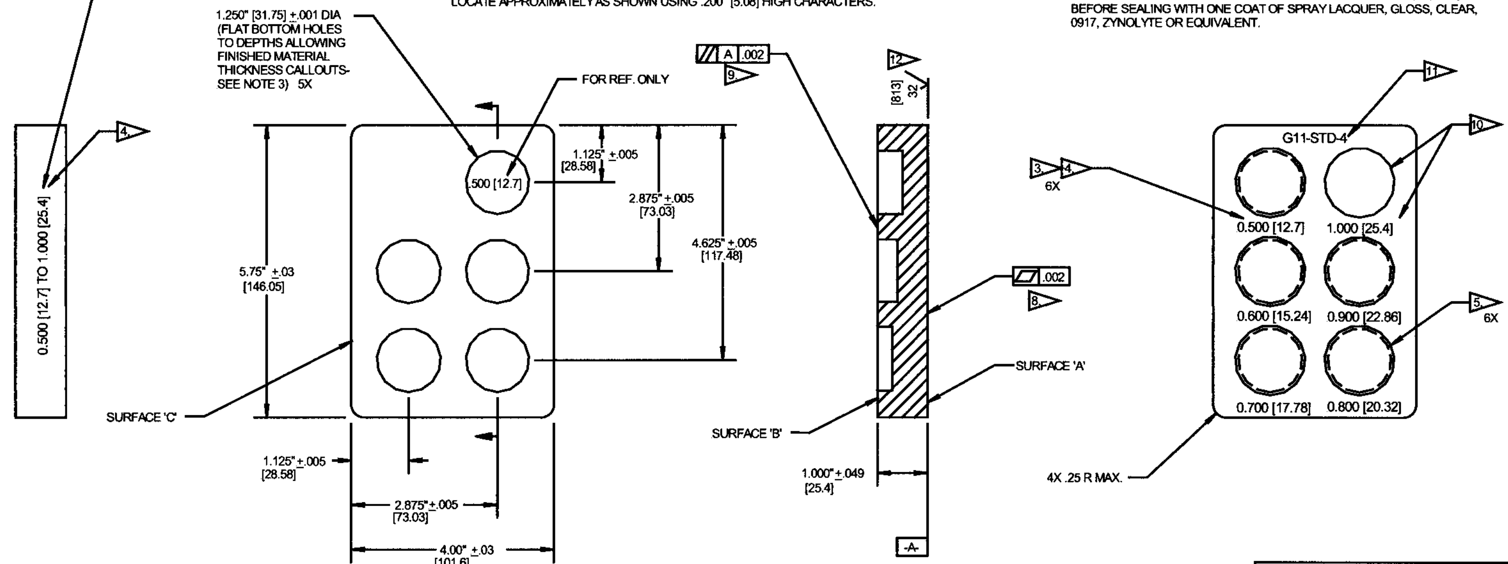
9. SURFACE 'B' TO BE PARALLEL TO SURFACE 'A' WITHIN .002 [.051]. THIS PARALLELISM IS A TOLERANCE ZONE DEFINED BY TWO PLANES PARALLEL TO SURFACE 'A' WITHIN WHICH THE LINE ELEMENTS OF THE SURFACE MUST LIE.

10. IN THIS AREA THREE MEASUREMENTS SHOULD BE TAKEN AND ALL SHOULD BE WITHIN TOLERANCE CALLED OUT ON THE PART. THE AVERAGE OF THESE MEASUREMENTS IS WHAT SHOULD BE MARKED ON THE PART.

11. MARK PART NUMBER WITH INK USING LEROY LETTERING GUIDE OR EQUIVALENT. LOCATE APPROXIMATELY AS SHOWN USING .200" [5.08] HIGH CHARACTERS.

NOTES:

1. MATERIAL: G11 PHENOLIC FABRICATED IN ACCORDANCE WITH Mil-I-24768/3, TYPE GEB, 1.000" [12.7] THK.
2. DIMENSIONS ARE IN INCHES AND NUMBERS IN [ ] ARE MILLIMETERS.
3. THE VALUES SHOWN REPRESENT THE THICKNESS OF MATERIAL LEFT IN THE FLAT BOTTOM HOLES AFTER THE MACHINING PROCESS WITHIN ±.001 [.025].
4. MARK DESIGNATIONS WITH INK USING LEROY LETTERING GUIDE OR EQUIVALENT. LOCATE APPROXIMATELY AS SHOWN USING .200" [5.08] HIGH CHARACTERS.
5. MARK A 1.375" [34.93] DIA CIRCLE CENTERED AROUND EACH FLAT BOTTOM HOLE ON THE FLAT SIDE WITH INK USING TEMPLATE. (USE INK THAT WILL NOT SMEAR WHEN SPRAYED WITH FINISH COAT).
6. AFTER MARKING DESIGNATIONS AND CIRCLES LET INK DRY OVERNIGHT BEFORE SEALING WITH ONE COAT OF SPRAY LACQUER, GLOSS, CLEAR, 0917, ZYNOLYTE OR EQUIVALENT.



TOLERANCE METRIC EQUIVALENTS	
INCHES	MILLIMETERS
.001	[.025]
.005	[.127]
.030	[.762]
.049	[1.24]

G11 STANDARD - 1.000"
Drawing No. G11-STD-4
Scale: 1/2 Date: 10-10-00
PREPARED UNDER JURISDICTION OF ATA/IATA/SAE COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE
PREPARED BY: CACRC INSPECTION TASK GROUP

4. NOTES

4.1 **Keywords:** Non-destructive inspection, composite laminate, reference standards, NDI.

PREPARED UNDER THE JURISDICTION OF ATA/IATA/SAE
COMMERCIAL AIRCRAFT COMPOSITE REPAIR COMMITTEE (CACRC)

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APPENDIX C

Test Specimen Matrix and Equipment Set-up Procedure for MAUS Inspections of Solid Composite Laminate Standards

Solid Composite Laminate Test Specimen Matrix

1. MAUS backdrill hole reference standard (carbon material with 50 mil thickness steps from 0.050" to 0.500" th).
2. Fiberglass Plate – 12"L X 12" H X 1" th plate with flat-bottomed holes ranging from 0.010" th to 0.755" th.
3. Carbon Fabric Plate - 12"L X 12" H X 1" th plate with flat-bottomed holes ranging from 0.010" th to 0.900" th.; made from plain weave BMS 8-212
4. G11 Phenolic Block Set #1 – set of four (5.75"L X 4" H) plates with flat-bottomed holes ranging from 0.010" th to 1" th.
5. G11 Phenolic Plate – single specimen (12"L X 12" H X 1" th) with flat-bottomed holes ranging from 0.010" th to 1" th.
6. Boeing ST8870 Carbon Fabric Step Wedge Set #1 – set of three wedge specimens (15" L X 1.5" H) made from BMS 8-212 plain weave carbon graphite cloth; 20 inspection sites with thicknesses ranging from 0.008" to 0.250" (this is Boeing's set)
7. Boeing ST8870 Carbon Fabric Step Wedge Set #2 – set of three wedge specimens (15" L X 1.5" H) made from BMS 8-212 plain weave carbon graphite cloth; 20 inspection sites with thicknesses ranging from 0.008" to 0.250" (this is Sandia Labs' set)
8. Boeing ST8871 Uniaxial Carbon Tape Step Wedge Set #1 – single specimen (10" L X 1" H) made from BMS 8-276 uniaxial carbon graphite tape; thickness ranges from 0.050" to 1" (this is Boeing's specimen)
9. Boeing ST8871 Uniaxial Carbon Tape Step Wedge Set #2 – single specimen (10" L X 1" H) made from BMS 8-276 uniaxial carbon graphite tape; thickness ranges from 0.050" to 1" (this is Sandia Labs' specimen)
10. Boeing ST8870 Fiberglass Step Wedge Set #1 - set of three wedge specimens (15" L X 1.5" H) made from fiberglass fabric; 20 inspection sites with thicknesses ranging from 0.005" to 0.126" (this is Boeing's set)
11. Boeing Crowsfoot Carbon Fabric Step Wedge Set # 1 – single wedge specimen (14" L X 2" H) made from BMS 8-212 four harness crowsfoot weave carbon graphite cloth; 8 inspection sites with thicknesses ranging from 0.023" to 0.297"
12. Boeing Crowsfoot Carbon Fabric Step Wedge Set # 2 – single wedge specimen (14" L X 2" H) made from BMS 8-212 four harness crowsfoot weave carbon graphite cloth; 8 inspection sites with thicknesses ranging from 0.022" to 0.294"
13. Boeing Carbon Fabric Step Wedge – single wedge specimen (15" L X 2" H) made from BMS 8-212 plain weave carbon graphite cloth; 7 inspection sites with thicknesses ranging from 0.253" to 0.828"
14. NWA Carbon Fabric Step Wedge #1 – single wedge specimen (22" L X 6" H) that was debulked during fabrication; 11 inspection sites with thickness ranging from 0.018" to 0.112"
15. NWA Carbon Fabric Step Wedge #2 – single wedge specimen (22" L X 6" H) that was not debulked during fabrication; 11 inspection sites with thickness ranging from 0.018" to 0.115

16. NWA Carbon Fabric Step Wedge #3 – single wedge specimen (22" L X 6" H) with unknown fabrication process; 11 inspection sites with thickness ranging from 0.018" to 0.110""

<u>MAUS Test Set-Up</u>

- Inspect parts using resonance mode.
- Null the transducer in the air. Maintain a single null point for all inspections so that the color codes can be compared from one specimen to another.
- Inspect areas of common thickness across as many of the 15 samples as possible. For example, 0.100" should be a common thickness among the specimens. Inspect at thicker and thinner regions as well even if it's not possible to inspect the full set the specimens at a particular thickness.
- Inspect a cluster of thicknesses, for example 0.116", 0.100", and 0.084", in order to characterize the color variations stemming from slight thickness changes. This will help us quantify the effects of small response differences between G11 and existing OEM standards.
- If the area permits, inspect a single thickness several times on a single specimen to accumulate statistical variation data. Note: since the MAUS system is scanning, it will inspect most of each thickness area (within probe deployment limits). Most of the specimens are not large enough to allow for much statistical sampling. We will, however, be able to study differences in similar standards for the ones where we have more than one set (i.e. matching items (4) and (5) and matching items (6) and (7) listed above).
- Inspect all of the thicknesses in the United Airlines specimens (specimen #'s 13-15) and compare with, at least, the G11 (specimen # 4) and Boeing Carbon Graphite step wedges (specimen # 6). This is a critical item when determining the amount of response variation will exist in the field over similar materials.

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